N077719

SATURN SUSTAINING ENGINEERING REPORT

SPACE TUG

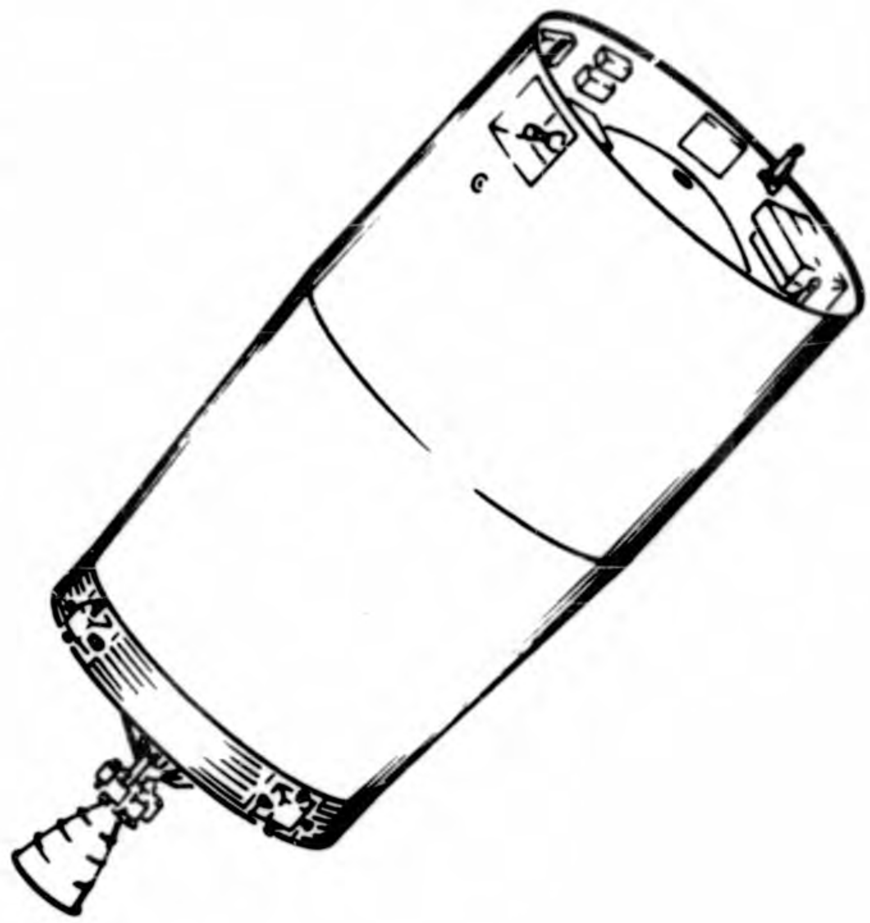
POINT DESIGN STUDY

FINAL REPORT

VOLUME III
DESIGN DEFINITION
PART - 2

PREPARED FOR
GEORGE C. MARSHALL
SPACE FLIGHT CENTER

SD72-SA-0032



FEBRUARY 11, 1972

358

Space Division
North American Rockwell

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SD72-SA-0032

SPACE TUG POINT DESIGN STUDY FINAL REPORT

VOLUME III

DESIGN DEFINITION

PART - 2

INSULATION, METEOROID PROTECTION, STRUCTURES, MASS PROPERTIES, GSE, RELIABILITY & SAFETY

FEBRUARY 11, 1972

Prepared for

George C. Marshall Space Flight Center

Approved by

R. Schwartz

Chief Program Engineer

FOREWORD

The final report on the Tug Point Design Study was prepared by the North American Rockwell Corporation through its Space Division for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with SA 2190 and Contract No. NAS 7-200.

The study effort described herein was conducted under the direction of NASA MSFC Study Leader, Mr. C. Gregg. The report was prepared by NR-SD, Seal Beach, California under the direction of Mr. T. M. Littman, Study Manager. The study results were developed during the period from 4 November 1971 through 11 February 1972 and the final report was submitted in February of 1972.

Valuable guidance and assistance was provided throughout the study by the following NASA/MSFC personnel:

C. Gregg - Study Leader
S. Denton - Structures
A. Willis - Avionics
J. Sanders - Propulsion
R. Nixon - Thermal Protection
A. Young - Flight Performance
R. L. Klan - Cost

The complete set of volumes comprising the report includes:

I Summary
II Operations, Performance, and Requirements
III Design Definition

Part 1 - Propulsion and Mechanical Subsystems, Avionic Subsystems, Thermal Control, and Electrical Power Subsystem

Part 2 - Insulation Subsystems, Meteoroid Protection, Structures, Mass Properties, Ground Support Equipment, Reliability, and Safety

IV Program Requirements
V Cost Analysis

This part of Volume III contains detailed descriptions of the Tug insulation, meteoroid protection, primary structure and ground support equipment, along with the technical assessment leading to the concept selection. Also included are the Tug mass properties, and the reliability and safety assessments.

ABSTRACT

The primary objective of the Tug Point Design Study was to verify through detail design and analysis the performance capability of a baseline design to deliver and retrieve payloads between 100 nautical miles/28.5 degrees inclination and geosynchronous. The Tug as groundruled for the study, is ground-based, reusable for 20 mission cycles, and is shuttled to and from low earth orbit by an Earth Orbital Shuttle (EOS) with a 65,000 pound payload capability. A 1976 state-of-the-art also was groundruled for the investigations.

The results of the effort show that the baseline concept can be designed to meet the target performance goals. Round trip payload capability to geosynchronous orbit is 3720 pounds; 720 pound margin over the established goal.

The design analysis performed to ascertain the Tug propellant mass fraction encompassed definition of the vehicle primary structure, thermal control, meteoroid protection, propulsion and mechanical subsystems, and avionics including power generation and distribution.

Graphite-epoxy composite material was determined to be feasible for Tug use and resulted in considerable weight savings. The concept of employing the primary load-carrying outer shell as a multi-function element integrating the meteoroid shield and insulation purge bag requirements is also feasible and enhances design simplicity. In addition, the use of a dual-mode pressure schedule during boost to orbit when applied loads are highest resulted in minimum tank weight. This, combined with an integrated gaseous O_2/H_2 auxiliary propulsion for stability and control, main tanks prepressurization, and fuel cell usage yield a minimum weight and operationally simple system.

Reliability and Safety analyses verified that no single failure of a component would result in a critical or unsafe condition. This was accomplished employing redundancy as required, notably in propulsion subsystems valving and attitude control components.

Program requirements were developed to verify the feasibility, producibility and operational capability of the point design. The results indicate that an "on-condition" maintenance approach similar to that used by commercial airlines and military operations would effectively serve Tug requirements.

Technology development study effort was concentrated on identifying the technologies needed for the baseline design. The more critical technologies requiring development include high performance engines, high performance insulation, large composite structures, and avionics.

A preliminary program development schedule was structured summarizing the integrated activities necessary to support the Tug through design development, production, and ground and flight testing.

The cost analysis performed covered the five major cost categories of DDT&E, first unit production, SR&T, average flight maintenance and refurbishment, and flight test vehicle refurbishment.

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INTRODUCTION

The Space Tug is a high performance propulsion stage designed to operate as an orbital maneuvering stage launched by the two-stage Space Shuttle. Because of the nature of the Tug mission, performance capability is very sensitive to Tug mass fraction. This study was conducted to answer the questions "What Tug mass fractions are really achievable by 1980?", and "What level of technology effort is required in order to build a Tug having the high performance defined in NASA/MSFC's Study Plan (Reference 1)?". Both questions are discussed below.

BACKGROUND

Several pre-Phase A Tug/OOS (Orbit-to-Orbit Shuttle) studies have been conducted for NASA and USAF agencies with a wide variation in the mass fractions quoted. NR performed a reusable Space Tug study for NASA-MSC in 1970-71 (Reference 2) and both NR and MDAC evaluated OOS feasibility for SAMSO/Aerospace Corporation in 1971 (References 3, 4). Additionally, two European teams conducted Tug system studies for the European Space Agency (ELDO) during 1970-71 (References 5, 6). Investigations also have been accomplished by MSFC and Aerospace Corporation. These studies considered a wide variety of design concepts and autonomy limits, ground and space-based operational requirements, degree of reusability, unmanned and manned payload implications, single and multistages, and different technology bases.

Projected NASA and DOD missions for the 1980's and beyond demand a Tug designed for a high degree of reusability and operational flexibility to assure significant improvement in space flight economy. Furthermore, Tug design must be compatible with Shuttle orbiter cargo bay size, weight limitations, and environment. For a ground-based system, consideration also must be given to Shuttle transport of a mated Tug/Payload.

OBJECTIVES

This point design study had one primary aim which was to be verified by design detail and analysis; namely, that a reusable, ground-based Space Tug with an IOC target by about the end of 1979 (1976 state-of-the-art) can carry a 3000-pound round trip payload between orbits at 100 nautical miles/28.5 degrees inclination and geosynchronous. The key constraint was use of a Space Shuttle having a 65,000 pound orbital delivery capability. A minimum usable propellant mass fraction of 0.895 also was desired. Additional study objectives were to (1) define the necessary supporting research and technology (SR&T) activities and their associated funding, and (2) determine Tug development, first production, and maintenance/repair costs.

STUDY SCOPE

The detail design of an integrated system was performed for a baseline concept. The concept was derived from MSFC's Study Plan and NR-selected materials, fabrication techniques, and subsystems resulting from currently available data and new trade studies.

Concurrent with the baseline study, options were evaluated having the potential for improving Tug mass fraction and mission performance. Emphasis was placed on the areas of alternate materials and subsystem, flight mode and operational variations, and use of advanced technology.

The study logic of Figure 1.3-1 depicts the major functional activities and outputs of these activities. The analyses performed to satisfy study objectives can be subdivided into three inter-related major efforts which started at study outset and ran concurrently to completion. Initiation of these efforts at the same time was made possible by the large amount of technical data available from the data bank. System requirements and criteria definition and program support gave the design definition effort and input data necessary for realistic structural, mechanical, thermal, and avionics subsystems design taking into account reliability and safety requirements. The three major tasks formed an iterative loop to the extent that the short study schedule permitted. As the design of each component and subsystem evolved, the results were fed to the supporting activities which served to increase the depth of analysis and visibility of the overall system characteristics with each succeeding step. This approach also adapted itself to the timely establishment of performance sensitivities and development of potentially attractive subsystem concepts.

STUDY GUIDELINES

This section highlights those elements of the NASA Study Plan (Reference 1) which were most influential in directing the NR effort toward the achievement of the aforementioned objectives.

Key Assumptions and Guidelines

The items listed below provided the key design and operational drivers for the Tug:

1. 1976 Materials & Concepts Technology

2. Unmanned Design, Fail-Safe Operation

3. Reusable - Lifetime of 20 Missions

4. Ground Based - Refurbishment After Each Mission

5. 6-Day on Orbit Stay Time Unattached to Shuttle

6. Flight Between 100 n mi Circular, 28.5 degrees Inclination Geosynchronous orbit

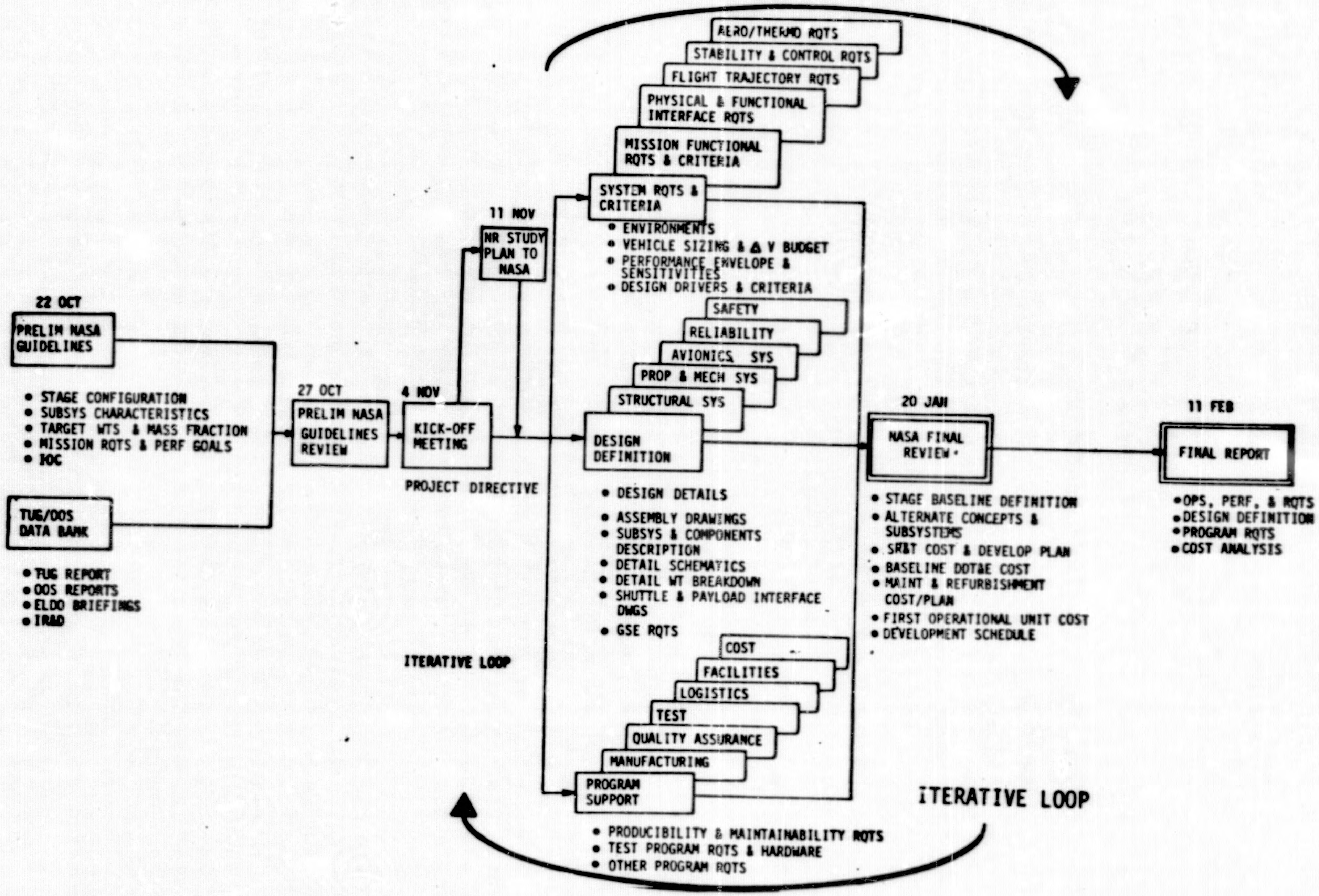

Figure 1.3-1 Top Level Study Logic

7. Payload Deliver/Retrieve Mixes in pounds

Baseline	3K/3K	
Alternates	0/4.16K 8.06K/0	Sizes Outer Shell Structure

8. Abort From Orbit Only & Propellant Dump/Inerting From Cargo Bay

9. Integrated MPS & APS Subsystems

Low vehile weight was a key design criterion due to the aforementioned performance objectives. Therefore, strong emphasis was given to the use of advanced materials and concepts deemed part of the 1976 technology base, but achievable without incurring severe cost penalties or high development risks. Fail-safe (FS) operations also provide for lower weight due to redundancy limitations (compared to the more demanding FO/FS requirements as employed in the OOS studies). However, FS does necessitate the highest practical component reliability to achieve an acceptable (over 0.9) mission success probability. Fail-safe is defined here as no failure modes which would cause an unsafe situation for the Shuttle or its crew, or destruction of the Tug payload. In the event of mission abort (limited to abort from orbit) while the Tug is still in the cargo bay, propellant dumping, tank inerting, and subsystems safing are required. These capabilities also are specified for normal re-entry and landing conditions to minimize hazards.

Unmanned design necessitates a high degree of subsystem/operational autonomy with ground support provided as emergency backup or when it yields weight and design simplicity advantages.

Reusability for 20 mission cycles (which may cover a period in excess of 3 years) can only be achieved in a practical cost-effective sense if airline-type servicing techniques are developed for Tug (as is planned for Shuttle). Strong attention must be given to assure a design compatible with this approach (accessibility, ease of inspection, and checkout).

The six-day orbital stay time affects cryogenic tankage protection and the total space exposure (for 20 missions) specifies meteoroid shielding requirements.

The baseline (3000 pound round trip) payload capability represents the most demanding from a performance (mass fraction) viewpoint. However, normal Shuttle ascent and descent carrying the Tug and the alternate payloads were employed to size the Tug outer shell structure, based on the flight load factors provided by MSFC for the study.

One additional sssumption agreed to between MSFC and NR, use of an integrated LOX/LH_2 propellant system for both main and auxiliary propulsion, provides design simplicity as well as weight and performance advantages.

Tug Baseline Concept

The NASA baseline configuration (Figure 1.4-1) which served as the starting point for this study is a single stage orbital propulsion system. It is limited to a maximum overall diameter of 15 feet and a maximum length of 35 feet, including Shuttle/Tug and Payload/Tug docking mechanisms. This vehicle is intended to separate from the Shuttle in orbit at 100 n mi/28.5 degrees inclination with a 3000-pound payload (15 ft x 25 ft) attached, ascend to geosynchronous orbit, deploy the up payload, retrieve a 3000-pound payload within 6000 n mi of the deployed payload, return to the near-vicinity of the Shuttle, redock, and return to earth. Payload center-of-gravity was defined as being at the geometric center of the 15 x 25 feet payload envelope.

The Tug has a non-integral tankage arrangement and is sized for a total propellant capacity of 56,394 pounds including 350 pounds of reserve plus allocations for reaction control/auxiliary propulsion (APS), fuel cell, residuals, and losses. The LH_2 tank has hemispherical bulkheads and a cylindrical section, whereas the LOX tank consists of two ellipsoidal bulkheads.

The docking systems are designed such that the active portion is left with the Tug in the Tug/payload interface and with the Shuttle in the Tug/Shuttle interface.

Other pertinent features are indicated on the profile. It should be noted that the Tug is attached at its aft end to the forward part of the orbiter cargo bay and thus is transported between Earth and orbit in an inverted attitude.

Tug Weight Targets

Table 1.4-1 lists the "bogey" weights provided by MSFC as design goals to assure meeting mass fraction requirements with the constraints of a 65,000 pound Shuttle capability and a 3000 pound Tug payload. No specific allocation was made for Tug-supportive hardware and fluids which remain in the EOS cargo bay. Instead, these were assumed to be contained within structure and other subsystems.

Structure includes all dry structure (docking mechanisms, meteoroid shield, outer shell, supports, thrust structure) and tankage subsystems. Thermal control includes cryogenic insulation, avionics cooling/heating hardware, and purge systems. Avionics contains GN&C, communications, data management, power generation and distribution, rendezvous and docking, and Tug electrical interfaces for ground and Shuttle and provisions for on-board checkout. Propulsion includes dry main engine, propellant feed, pressurization, fill/drain and vent/purge umbilicals, propellant dump, tank baffles/screens, APS thrusters/feed system/tanks, main engine actuators, and ullage venting control.

Non-usable fluids include propellant reserves, pressurant, thermal control fluids, and residuals. The main engine propellant bogey weight contains all propellant burned by the main engine during a nominal mission. APS propellant includes all burned attitude control and small delta-V translational maneuver requirements during a nominal mission. The miscellaneous fluids category

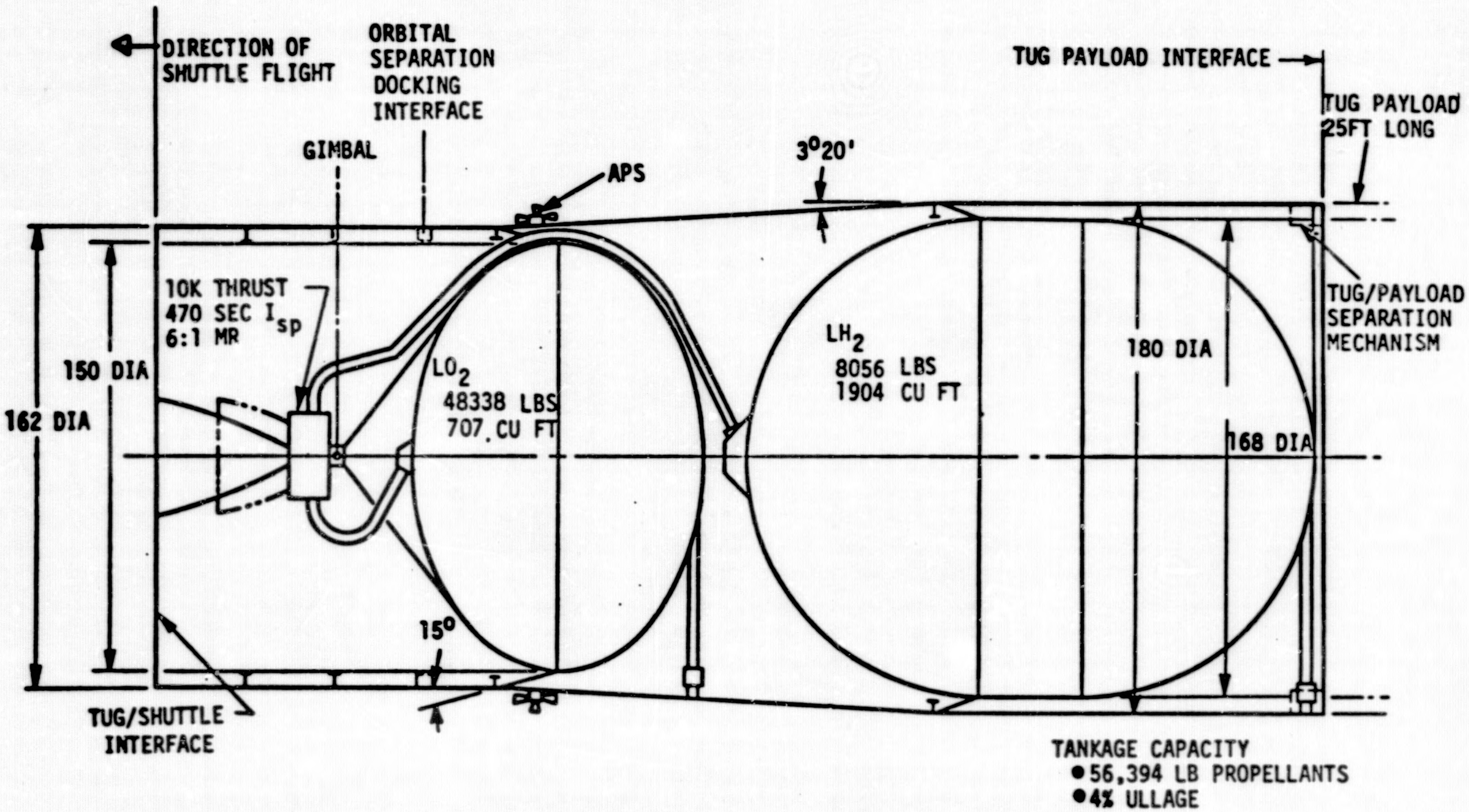

Figure 1.5-1 NASA Tug Baseline Concept & Sizing

contains all other unburned fluids (fuel cell, reactants, and vent/chilldown/start-stop losses). These have been numerically lumped together with non-usable fluids in the bogey weight table.

Table 1.4-1. Tug Bogey Weights

	Weight (lb)
Structure	2,552
Thermal Control	476
Avionics	1,011
Propulsion	1,057
Dry Weight	5,096
10% Contingency	510
Non-usable Fluids	842*
Burnout Weight	6,448
Usable Main Engine Propellant	55,148
Usable APS Propellant	404
Misc Fluids & Losses	--
Tug Flight Wt at Tug/EOS Separation	62,000
EOS Payload - Chargeable Interface Prov	--
Tug Gross Wt at EOS Liftoff	62,000
Gross EOS Payload at Liftoff	65,000
Mass Fraction, λ =	0.895
*Incl. 350 lb prop reserve	

INTEGRATED SYSTEM

The Space Tug may be structurally divided into four (4) basic sections:

1. The outer shell or unpressurized structure.
2. The propellant tanks and supports with their associated equipment or the pressurized structure.
3. The thrust structure and engine system.
4. Miscellaneous mounting provisions for various subsystem equipment (Avionics, APS, EPS, etc.)

As shown on the inboard profile Figure 1.5-1 the Tug is comprised of a shell structure within which the propellant tanks and engine system (thrust structure) are suspended. Wherever feasible the structure has been designed with a multi-purpose function in order to minimize the structural weight. This is most evident in the cylindrical outer shell where in addition to providing the primary structural load path, the shell functions as a purge bag and meteoroid shield. In every instance concerning the structure, the lightest weight system has been of primary importance.

The outer shell structure can be divided into the forward skirt, the intertank structure, the aft skirt and the EOS adapter. Each of these shells is a graphite epoxy skin over aluminum honeycomb core structure of varying thicknesses and shapes.

The forward skirt is cylindrical, 180 inches in diameter and 147.5 inches long. The shell is 3/8 of an inch thick. The aft end of the cylinder attaches to the forward end of the intertank structure at the LH_2 tank attach frame. The forward skirt is mechanically fastened at this joint.

At the forward end, the shell provides the interface with the payload and consequently there is a deep channel ring frame at this location. This frame is a graphite epoxy composite with gussets and web stiffeners, of the same material, as required. It is 16 inches deep at the interface plane. Payload docking and latching provisions are accommodated on this frame. Four additional stability ring frames are equally spaced on the forward skirt. The first frame aft of the forward interface is a larger frame, 6 inches deep, since it provides support for the docking probes. The remaining three frames are small stability frames, 1.5 inches deep. All of the frames are layed up from plies of graphite epoxy composite.

This shell is composed of 4 quarter panel subassemblies which are spliced with longitudinal strips of graphite epoxy composite (same as facing sheets). Several cutouts are provided in the shell to accommodate the antennas, umbilicals, and star tracker and horizon sensor field of views. Reinforcement is provided around the smaller cutouts by added plies of the facing sheet material while around the larger cutouts, channel-shaped intercostals between frames on each side of the opening provide the necessary stiffening. The heavy frame at the forward interface distributes the orbiter cargo bay attachment loads, and supports the payload attachment. These external fittings, which terminate in hemispherical ends, engage the orbiter cargo bay supports. Each fitting is loaded only in the tangential direction to the frame. Two fittings (Z-Z in orbiter) to carry vertical loads only are located on the side of the ring and one lower fitting reacts the lateral load (Y-Y in orbiter). The ring frame then distributes these loads to the shell structure.

The forward skirt accommodates eight panels of avionics equipment which is supported between the forward interface frame and the first frame aft of this. The panels are rectangular shaped 0.5 inch thick aluminum honeycomb with 0.020 inch thick aluminum skins. Two panels supporting the star tracker and communications require louvers on the back side of the panel to reject heat to the outer shell. Each panel does, however, require a heater to

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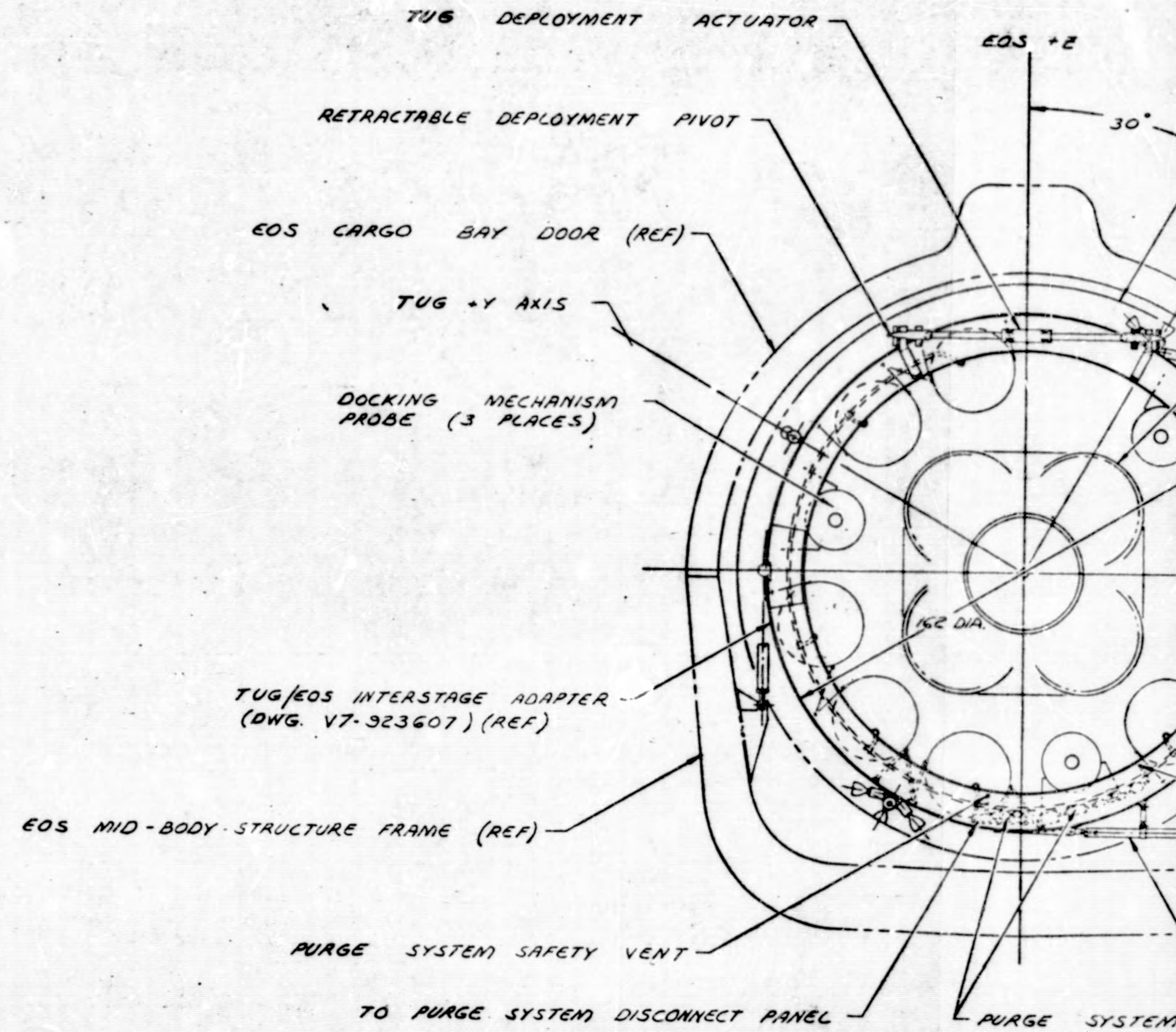
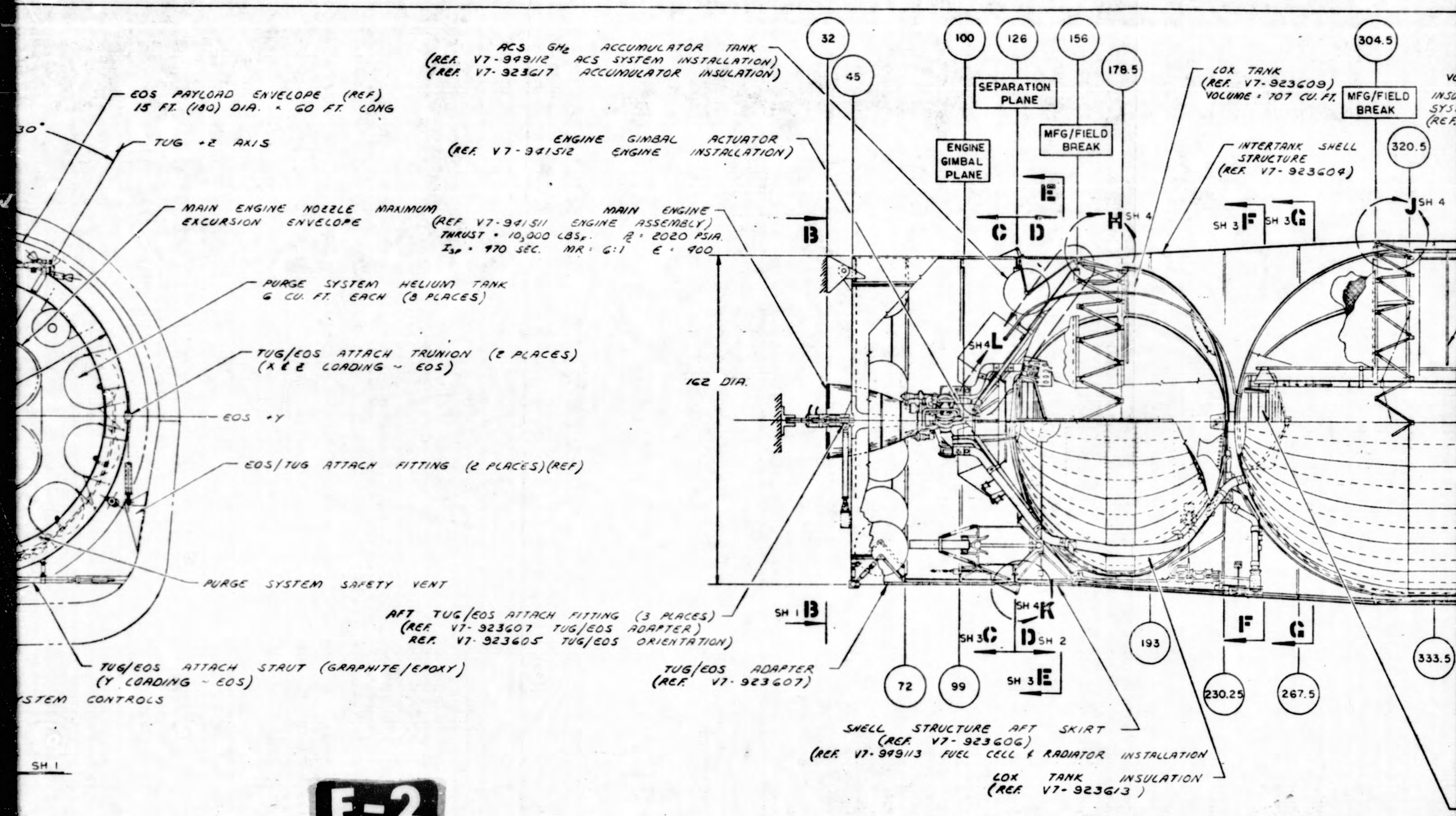


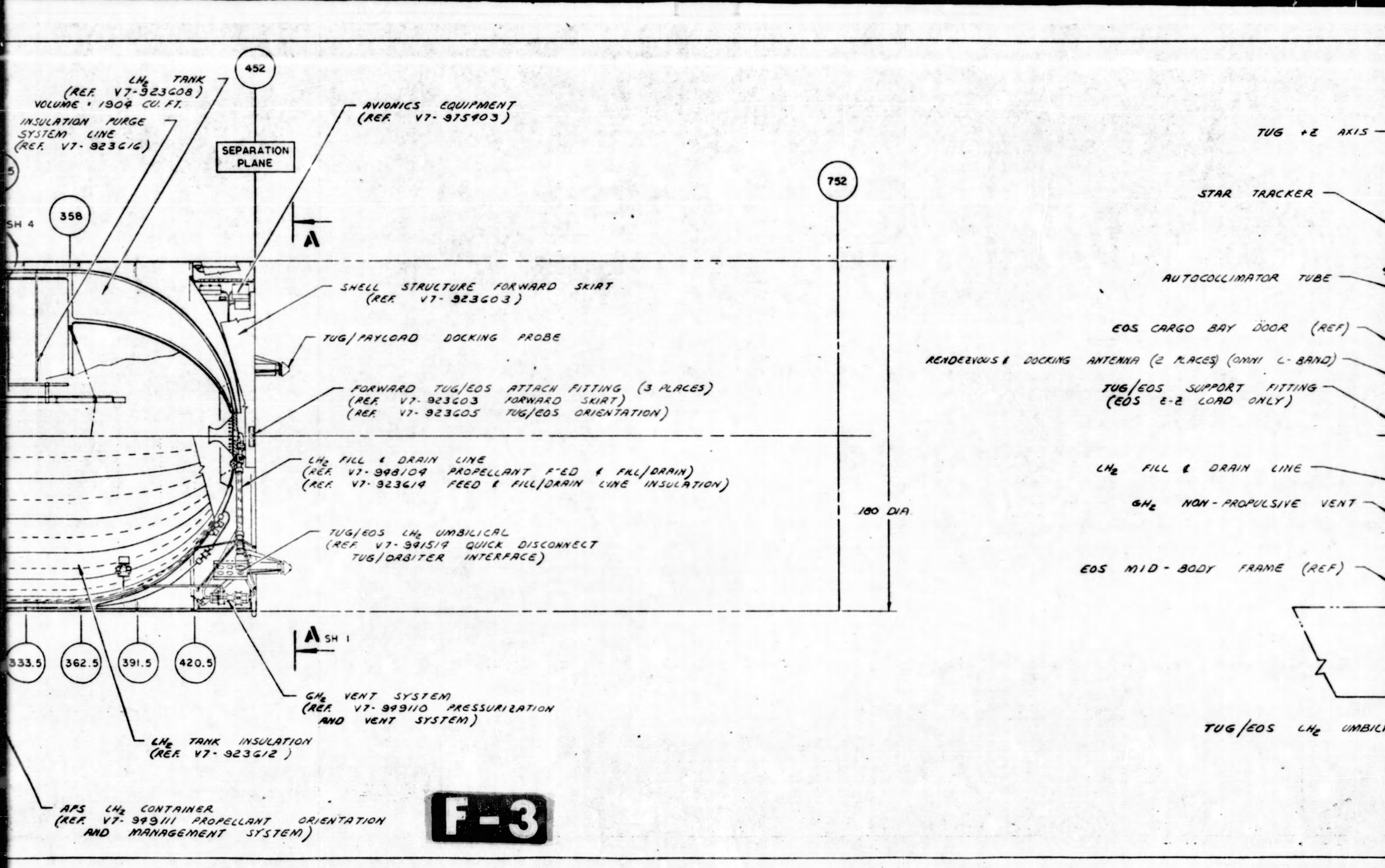
VIEW B - B SH 1

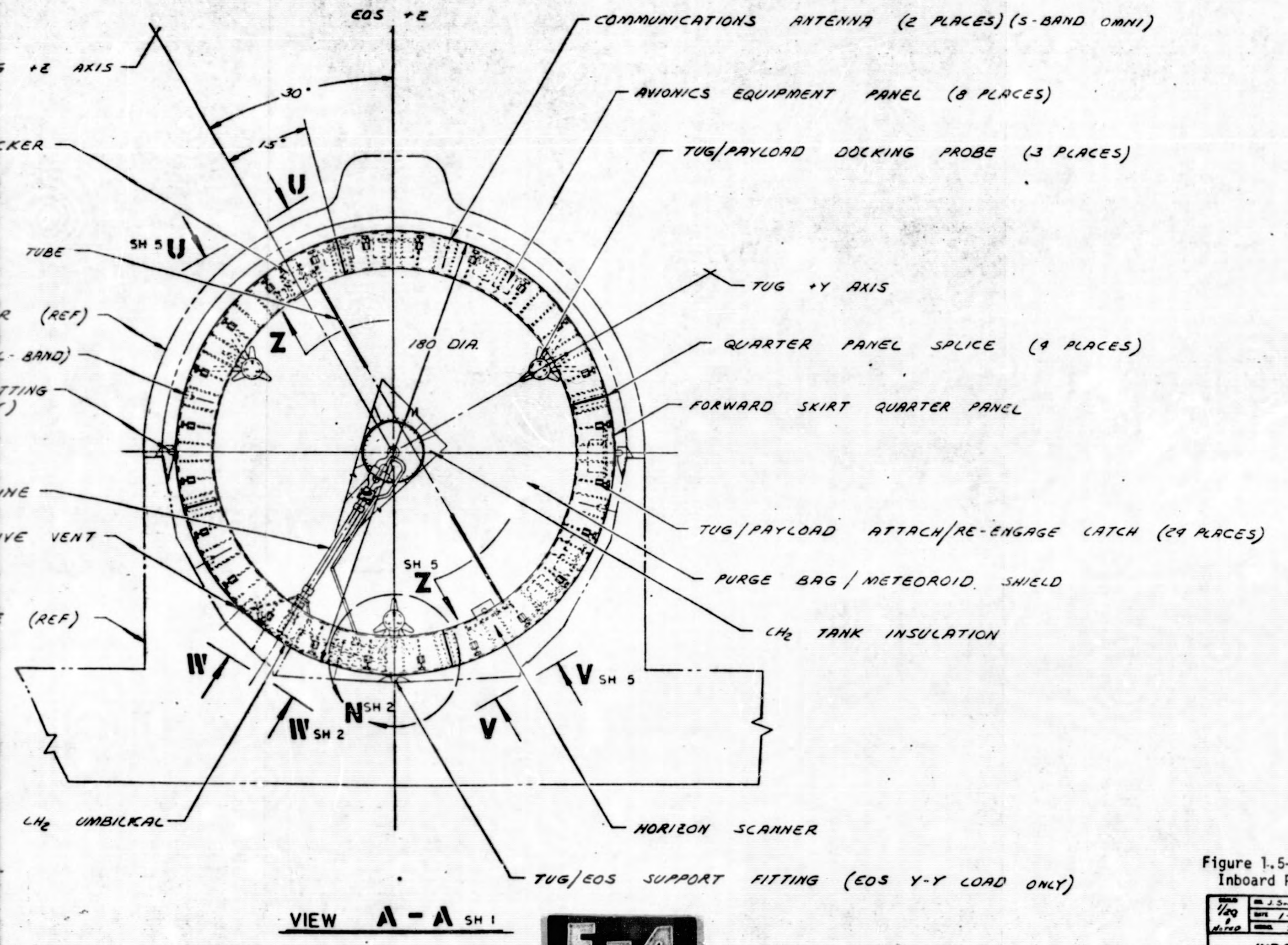
F-1



ACS GH_2 ACCUMULATOR TANK
(REF. V7-949112 ACS SYSTEM INSTALLATION)
(REF. V7-923617 ACCUMULATOR INSULATION)
EOS PAYLOAD ENVELOPE (REF)
15 FT. (180) DIA. × 60 FT. LONG
30°
TUG +Z AXIS
ENGINE GIMBAL ACTUATOR
(REF. V7-941512 ENGINE INSTALLATION)
MAIN ENGINE NOZZLE MAXIMUM EXCURSION ENVELOPE
MAIN ENGINE
(REF V7-941511 ENGINE ASSEMBLY)
THRUST = 10,000 LBSF. P_c = 2020 PSIA.
I_{SP} = 470 SEC. MR = 6:1 ε = 400
PURGE SYSTEM HELIUM TANK
6 CU. FT. EACH (8 PLACES)
TUG/EOS ATTACH TRUNION (2 PLACES)
(X & Z LOADING ~ EOS)
162 DIA.
EOS +Y
EOS/TUG ATTACH FITTING (2 PLACES)(REF)
PURGE SYSTEM SAFETY VENT
AFT TUG/EOS ATTACH FITTING (3 PLACES)
(REF. V7-923607 TUG/EOS ADAPTER)
REF. V7-923605 TUG/EOS ORIENTATION)
TUG/EOS ATTACH STRUT (GRAPHITE/EPOXY)
(Y LOADING ~ EOS)
TUG/EOS ADAPTER
(REF. V7-923607)
YSTEM CONTROLS
SH 1
32
45
100
126
156
178.5
304.5
320.5
SEPARATION PLANE
ENGINE GIMBAL PLANE
MFG/FIELD BREAK
LOX TANK
(REF. V7-923609)
VOLUME = 707 CU. FT.
MFG/FIELD BREAK
INTERTANK SHELL STRUCTURE
(REF. V7-923604)
B
C
D
E
SH 4 H
SH 3 F
SH 3 G
J SH 4
SH 4 L
SH 1 B
SH 4 K
SH 3 C
D SH 2
SH 3 E
F
G
72
99
193
230.25
267.5
333.5
SHELL STRUCTURE AFT SKIRT
(REF. V7-923606)
(REF. V7-949113 FUEL CELL & RADIATOR INSTALLATION
LOX TANK INSULATION
(REF. V7-923613)

F-2


452
LH_2 TANK
(REF. V7-923608)
VOLUME = 1904 CU. FT.
INSULATION PURGE SYSTEM LINE
(REF. V7-923616)
AVIONICS EQUIPMENT
(REF. V7-975403)
SEPARATION PLANE
TUG +Z AXIS
752
358
SH 4
STAR TRACKER
A
AUTOCOLLIMATOR TUBE
SHELL STRUCTURE FORWARD SKIRT
(REF. V7-923603)
TUG/PAYLOAD DOCKING PROBE
EOS CARGO BAY DOOR (REF)
RENDEZVOUS & DOCKING ANTENNA (2 PLACES) (OMNI C-BAND)
FORWARD TUG/EOS ATTACH FITTING (3 PLACES)
(REF. V7-923603 FORWARD SKIRT)
(REF. V7-923605 TUG/EOS ORIENTATION)
TUG/EOS SUPPORT FITTING
(EOS Z-Z LOAD ONLY)
LH_2 FILL & DRAIN LINE
(REF. V7-998104 PROPELLANT FEED & FILL/DRAIN)
(REF. V7-923614 FEED & FILL/DRAIN LINE INSULATION)
LH_2 FILL & DRAIN LINE
180 DIA.
GH_2 NON-PROPULSIVE VENT
TUG/EOS LH_2 UMBILICAL
(REF. V7-991519 QUICK DISCONNECT TUG/ORBITER INTERFACE)
EOS MID-BODY FRAME (REF)
A SH 1
333.5
362.5
391.5
420.5
GH_2 VENT SYSTEM
(REF. V7-999110 PRESSURIZATION AND VENT SYSTEM)
TUG/EOS LH_2 UMBILIC
LH_2 TANK INSULATION
(REF. V7-923612)
APS LH_2 CONTAINER
(REF. V7-999111 PROPELLANT ORIENTATION AND MANAGEMENT SYSTEM)
F-3



F-4

Figure 1.5-1
Inboard Profile Tug (Sheet 1 of 5)

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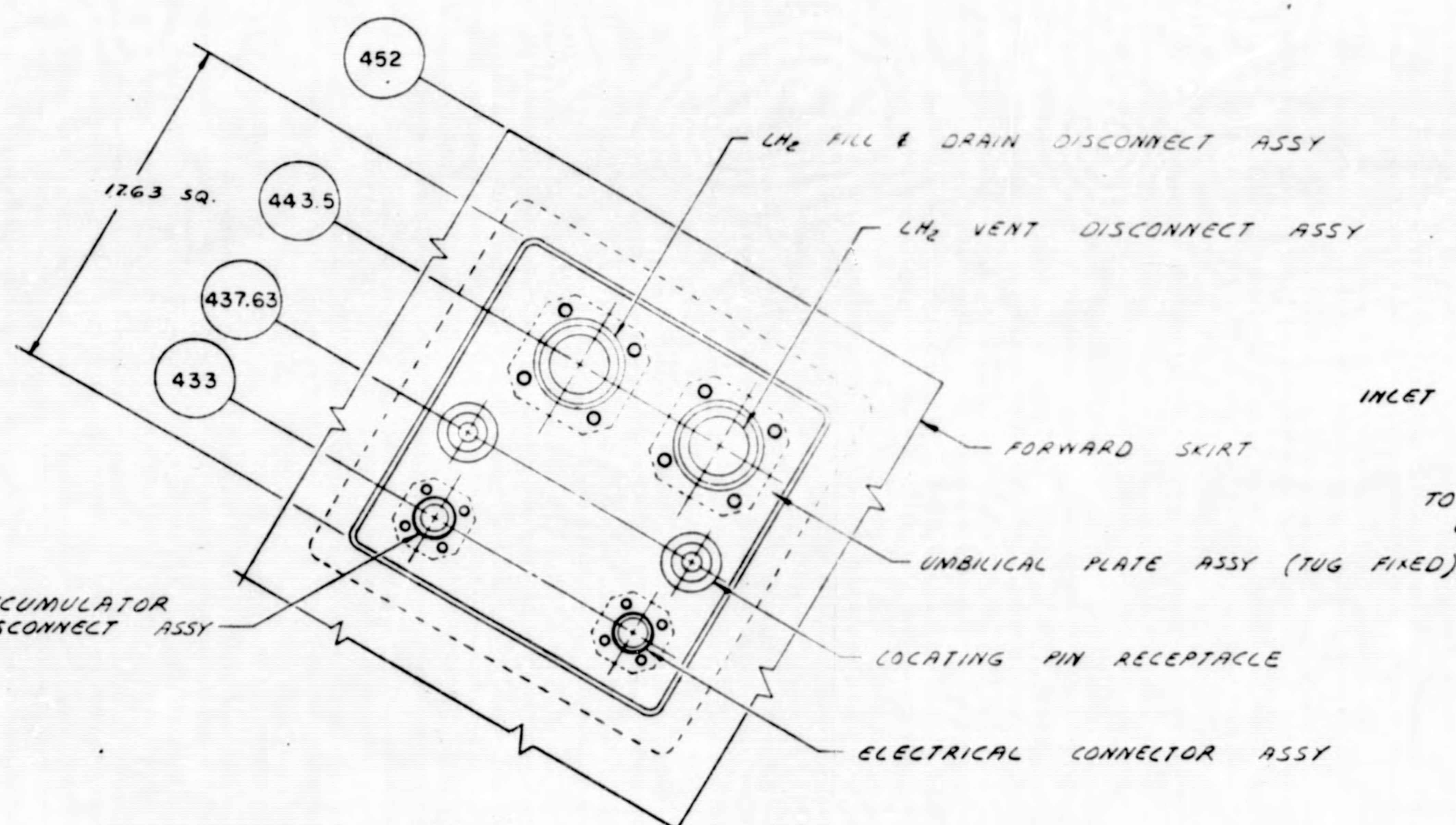
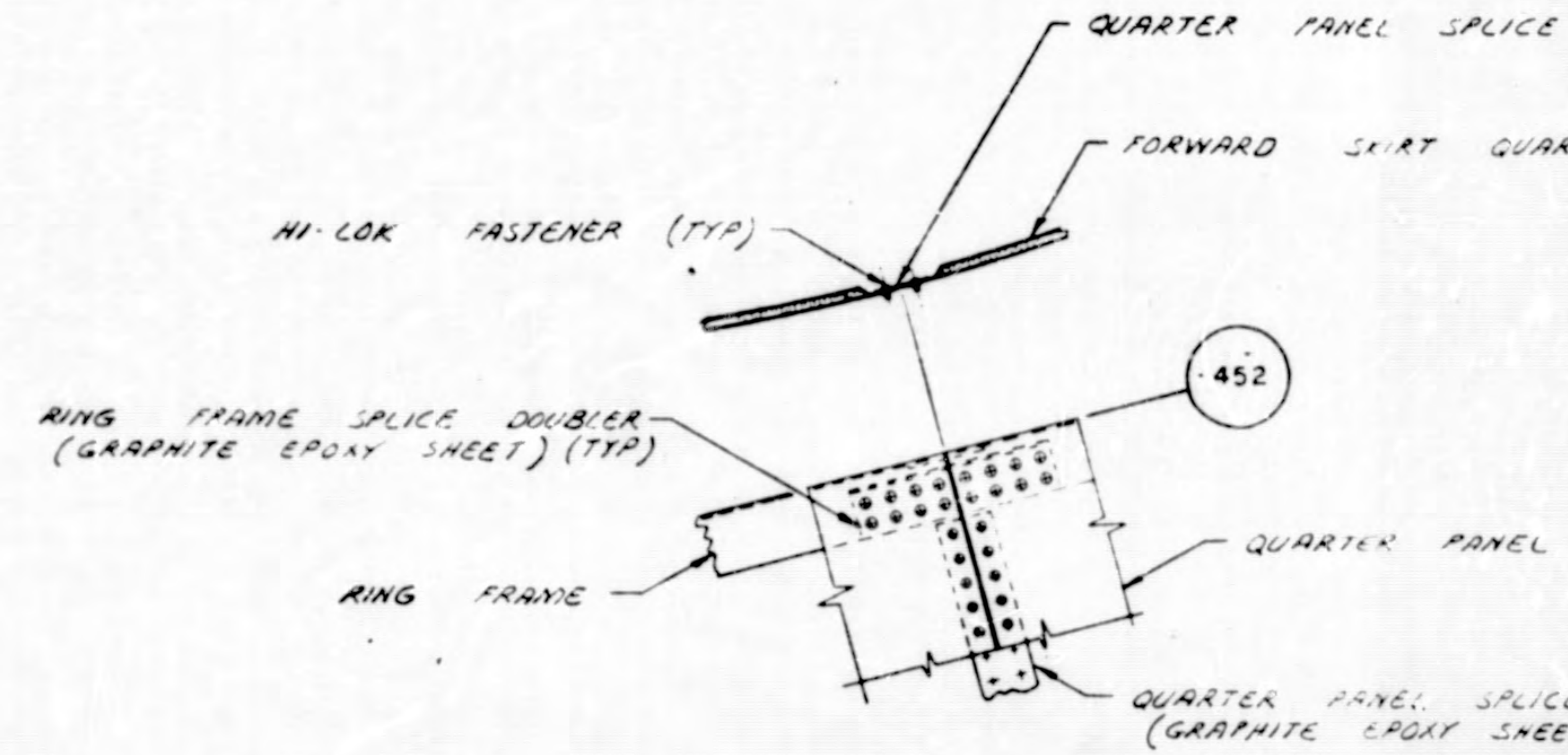
INBOARD PROFILE TUG		V7-923602 SH 1 OF 5



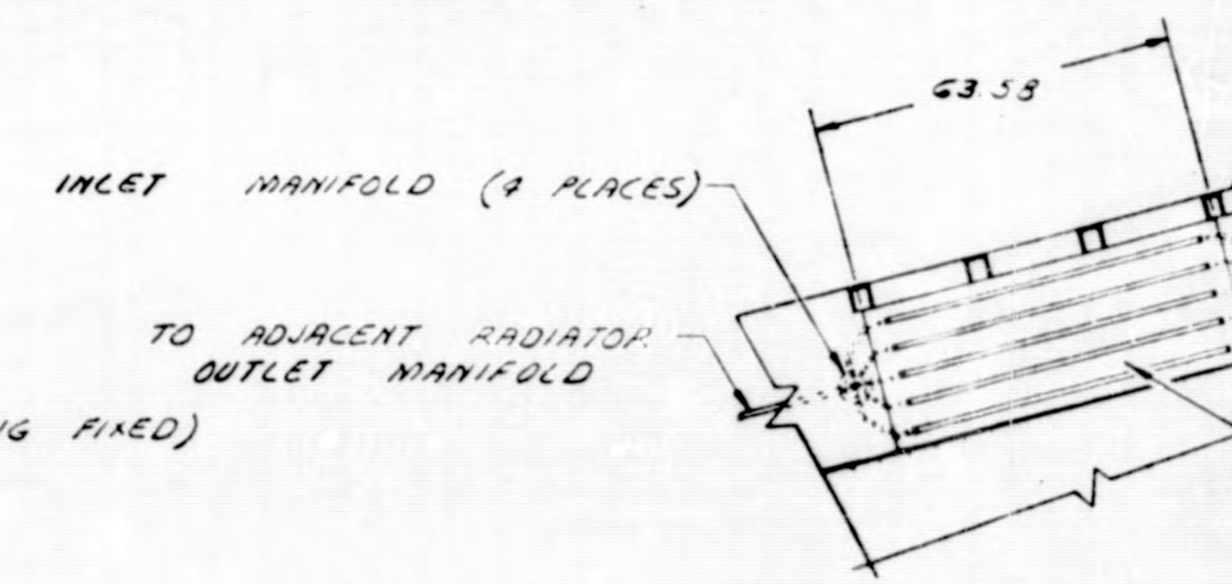
VIEW W - W SH 1

SCALE: 1/4



VIEW Y - Y SH 2

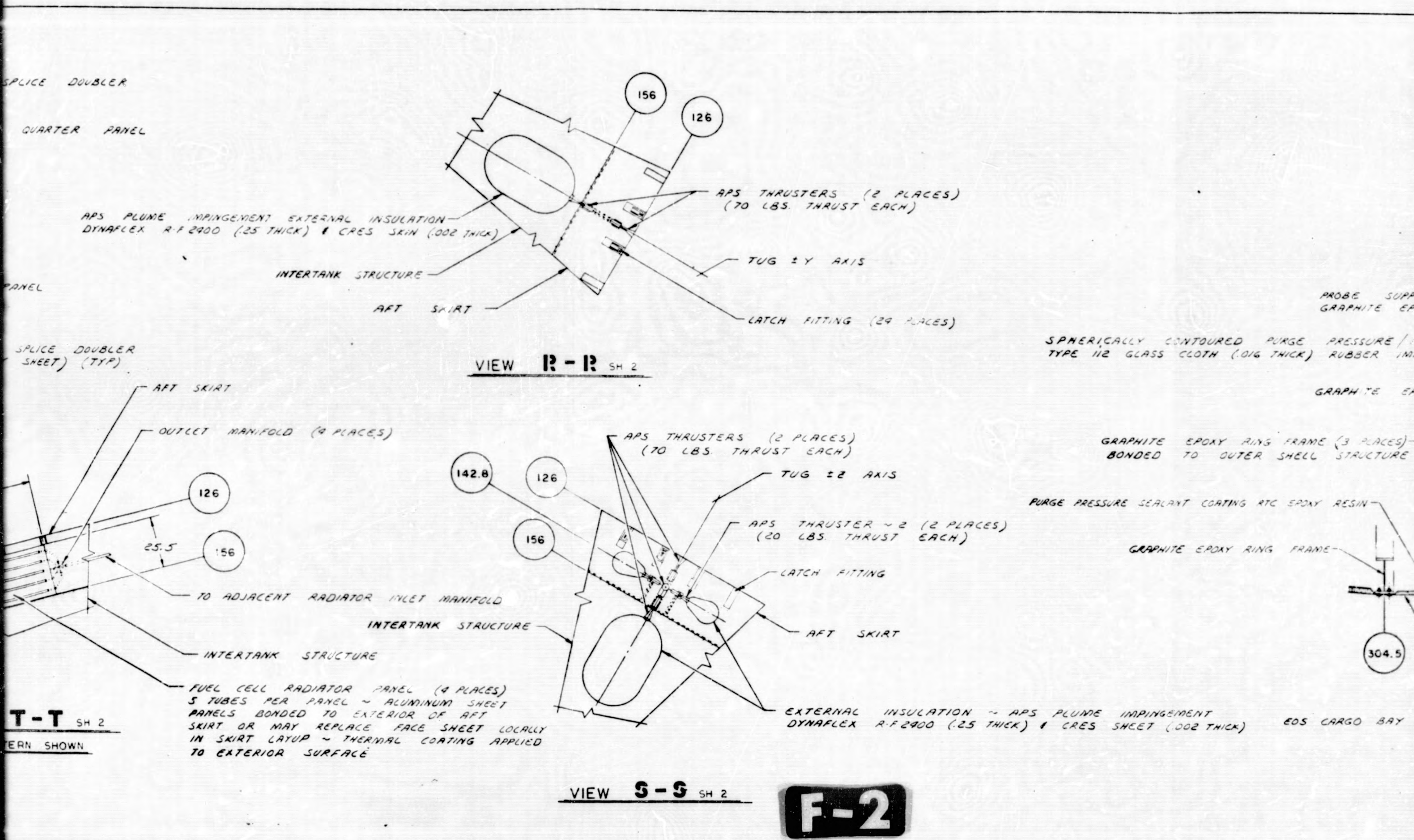
SCALE: 1/4



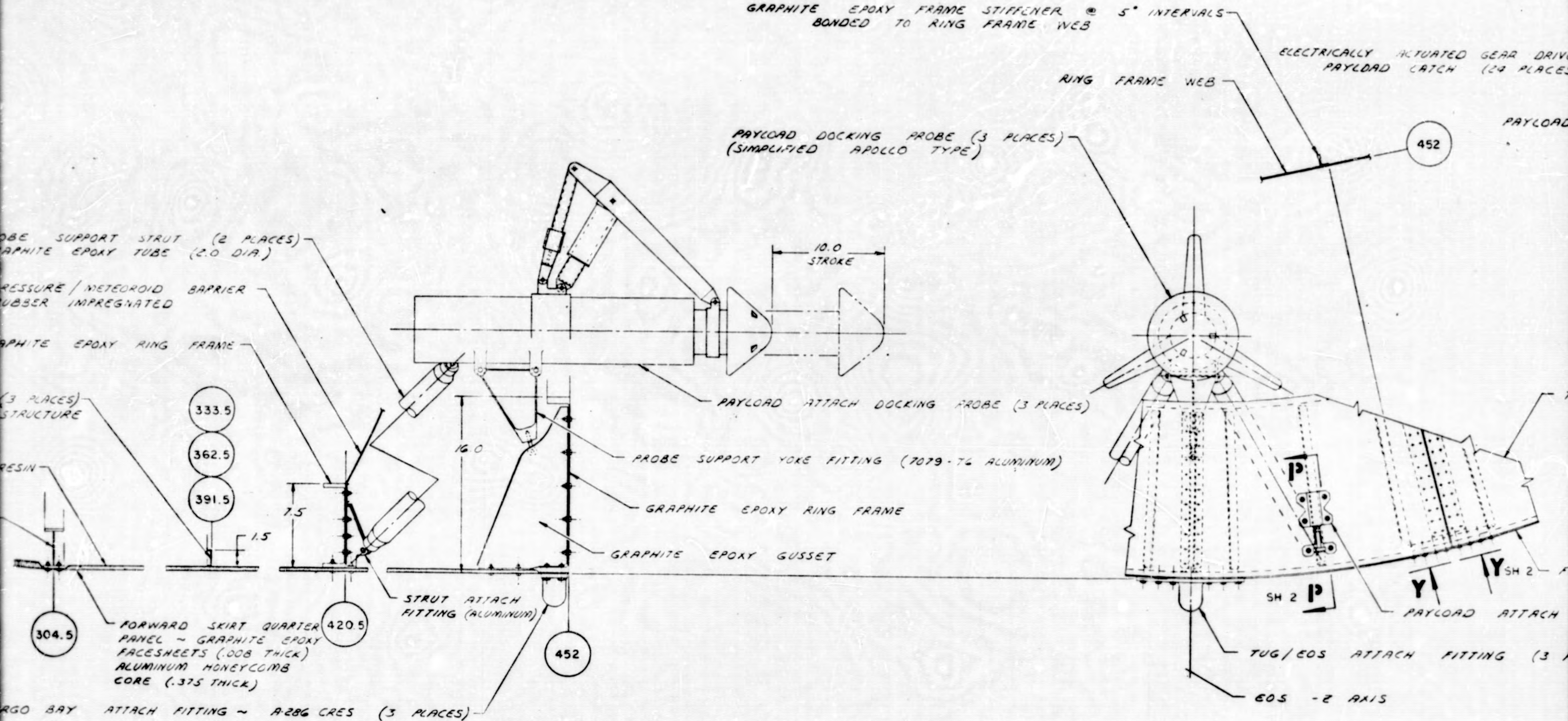
VIEW T - T

FLAT PATTERN SH

F-1


SPLICE DOUBLER
QUARTER PANEL
PANEL
SPLICE DOUBLER
SHEET) (TYP)
AFT SKIRT
OUTLET MANIFOLD (4 PLACES)
126
25.5
156
TO ADJACENT RADIATOR INLET MANIFOLD
INTERTANK STRUCTURE
FUEL CELL RADIATOR PANEL (4 PLACES)
5 TUBES PER PANEL ~ ALUMINUM SHEET
PANELS BONDED TO EXTERIOR OF AFT
SKIRT OR MAY REPLACE FACE SHEET LOCALLY
IN SKIRT LAYUP ~ THERMAL COATING APPLIED
TO EXTERIOR SURFACE
T-T SH 2
TERN SHOWN
156
126
APS THRUSTERS (2 PLACES)
(70 LBS THRUST EACH)
APS PLUME IMPINGEMENT EXTERNAL INSULATION
DYNAFLEX R-F 2400 (.25 THICK) & CRES SKIN (.002 THICK)
TUG ±Y AXIS
INTERTANK STRUCTURE
AFT SKIRT
LATCH FITTING (24 PLACES)
VIEW R-R SH 2
APS THRUSTERS (2 PLACES)
(70 LBS THRUST EACH)
142.8
126
156
TUG ±Z AXIS
APS THRUSTER ~ 2 (2 PLACES)
(20 LBS THRUST EACH)
LATCH FITTING
INTERTANK STRUCTURE
AFT SKIRT
EXTERNAL INSULATION ~ APS PLUME IMPINGEMENT
DYNAFLEX R-F 2400 (.25 THICK) & CRES SHEET (.002 THICK)
VIEW S-S SH 2
PROBE SUPPO
GRAPHITE EPO
SPHERICALLY CONTOURED PURGE PRESSURE / M
TYPE 112 GLASS CLOTH (.016 THICK) RUBBER IMP
GRAPHITE EPO
GRAPHITE EPOXY RING FRAME (3 PLACES)
BONDED TO OUTER SHELL STRUCTURE
PURGE PRESSURE SEALANT COATING RTC EPOXY RESIN
GRAPHITE EPOXY RING FRAME
304.5
EOS CARGO BAY

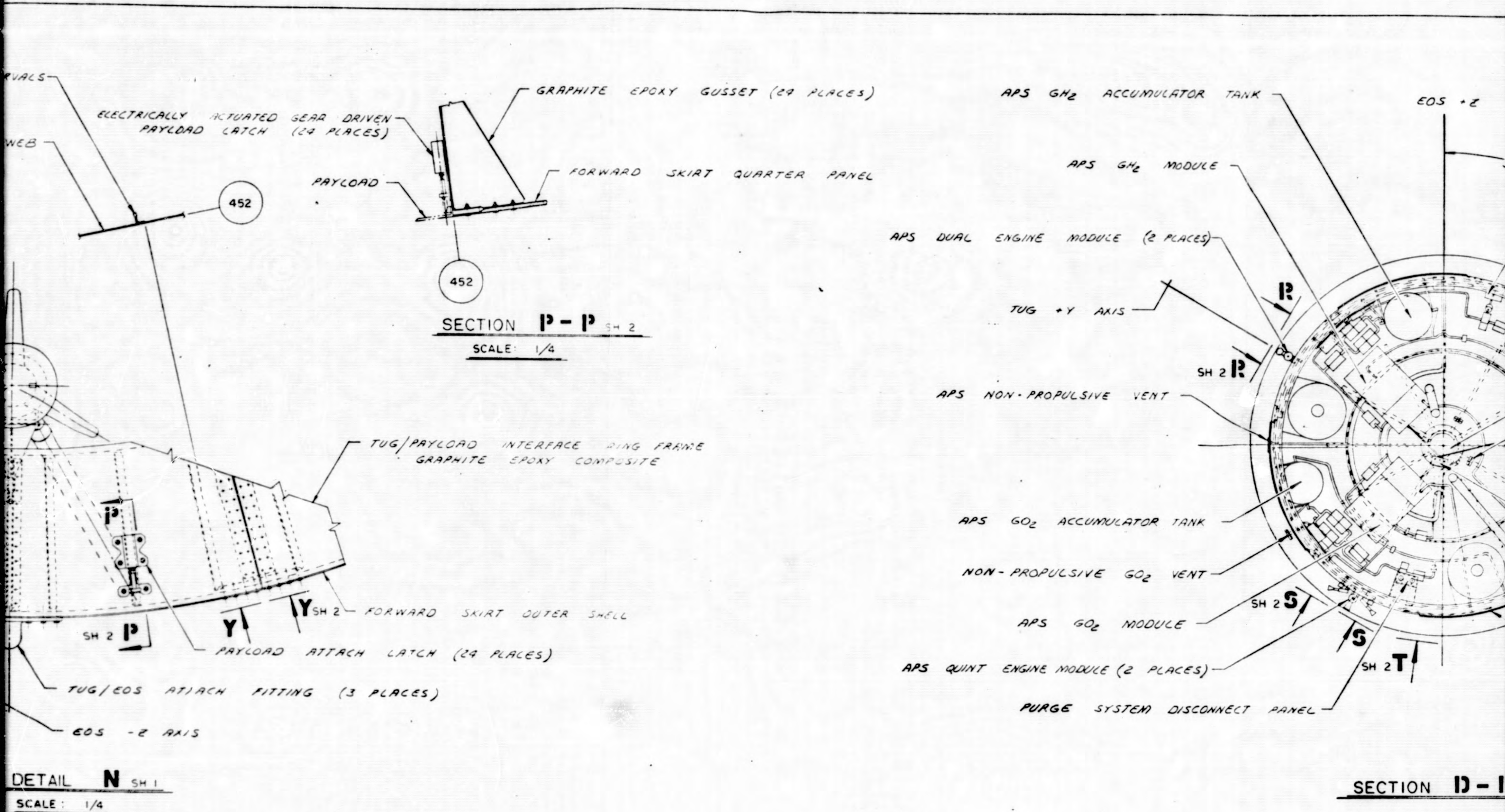
F-2



DETAIL N SH 1

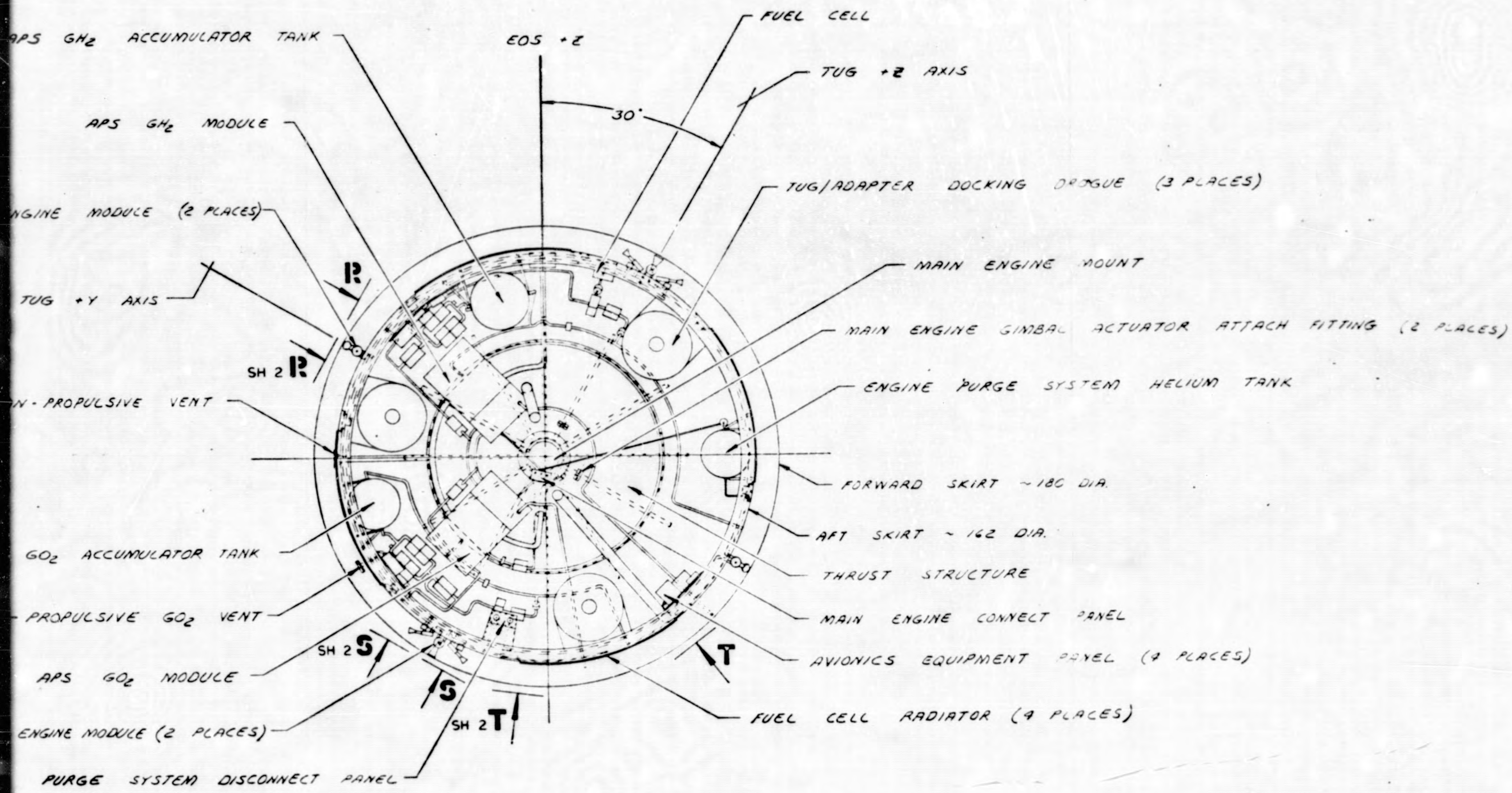
SCALE : 1/4

F-3



RVALS
ELECTRICALLY ACTUATED GEAR DRIVEN
PAYLOAD LATCH (24 PLACES)
WEB
452
GRAPHITE EPOXY GUSSET (24 PLACES)
PAYLOAD
FORWARD SKIRT QUARTER PANEL
452
SECTION P - P SH 2
SCALE: 1/4
TUG/PAYLOAD INTERFACE RING FRAME
GRAPHITE EPOXY COMPOSITE
P
SH 2 P
Y
Y SH 2 FORWARD SKIRT OUTER SHELL
PAYLOAD ATTACH LATCH (24 PLACES)
TUG/EOS ATTACH FITTING (3 PLACES)
EOS -Z AXIS
DETAIL N SH 1
SCALE: 1/4
APS GH_2 ACCUMULATOR TANK
EOS +Z
APS GH_2 MODULE
APS DUAL ENGINE MODULE (2 PLACES)
R
TUG +Y AXIS
SH 2 R
APS NON-PROPULSIVE VENT
APS GO_2 ACCUMULATOR TANK
NON-PROPULSIVE GO_2 VENT
SH 2 S
APS GO_2 MODULE
S
APS QUINT ENGINE MODULE (2 PLACES)
SH 2 T
PURGE SYSTEM DISCONNECT PANEL
SECTION D - D

F-4



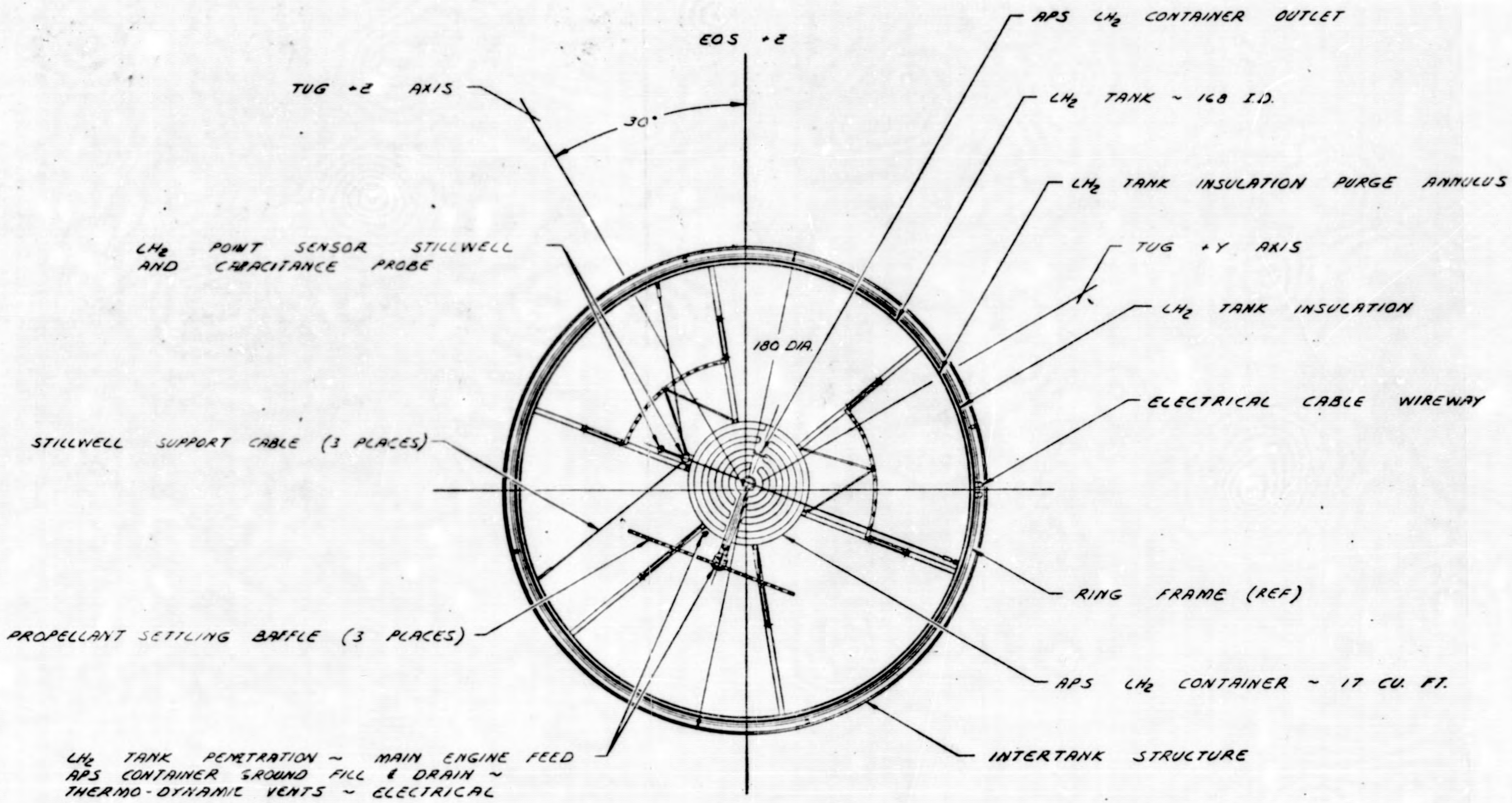
SECTION D - D SH 1

Figure 1.5-1
Inboard Profile Tug (Sheet 2 of 5)

SCALE 1/20	DR. J. S. / DATE 1-29-72 / MODEL TUG	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY CALIFORNIA
INBOARD PROFILE TUG		V7-923602 SH 2 OF 5

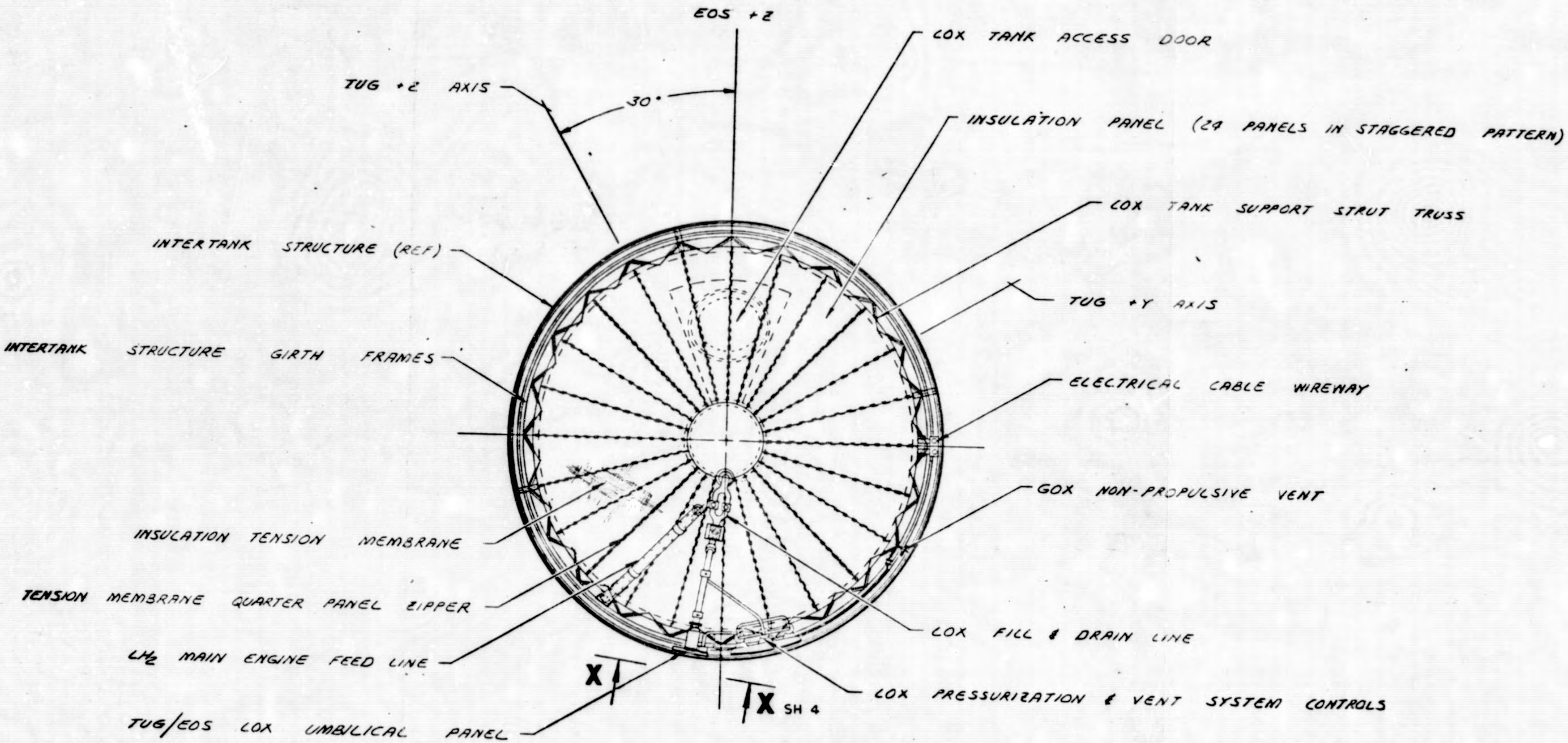
F-5

I-11, I-12



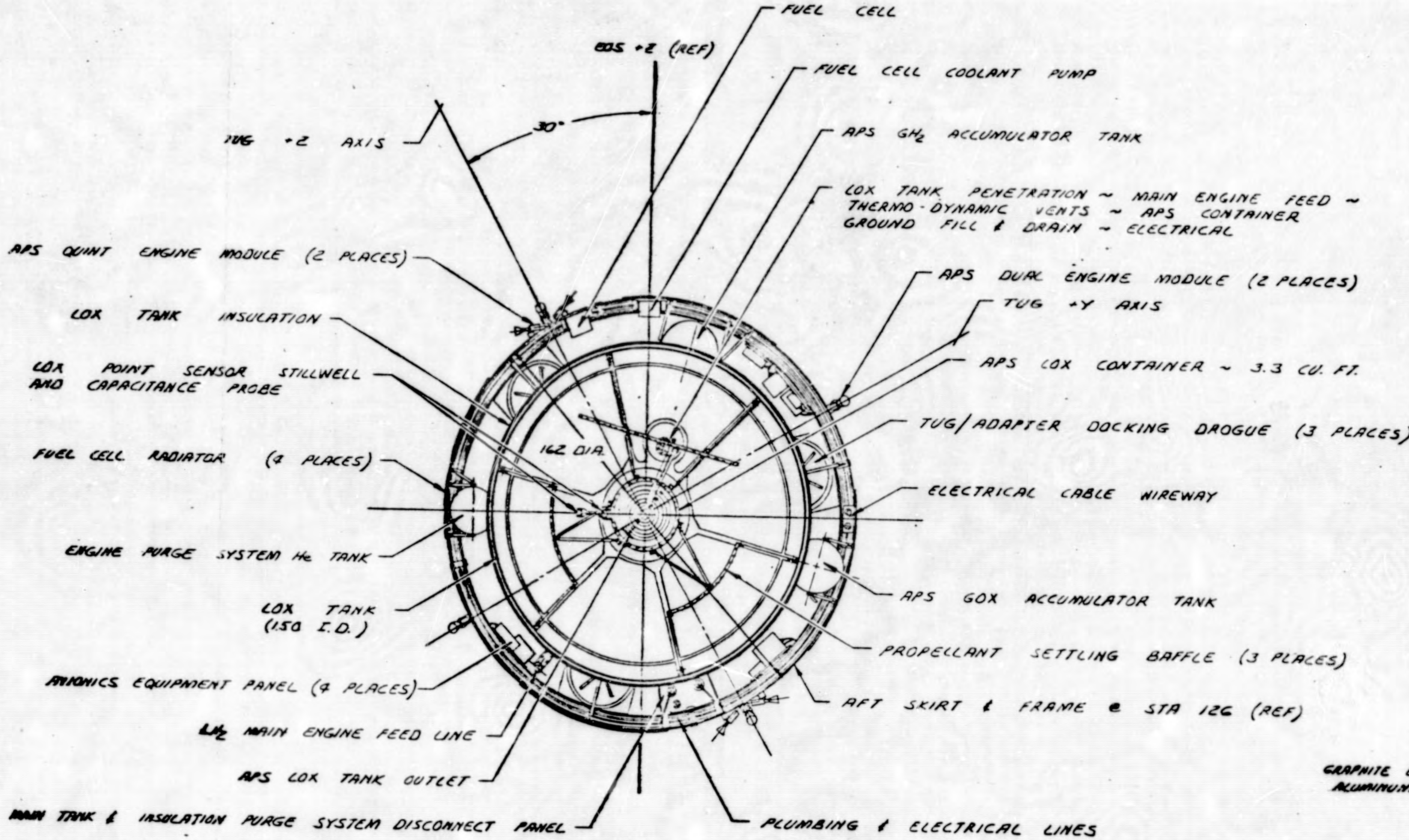
SECTION G - G SH 1

F-1

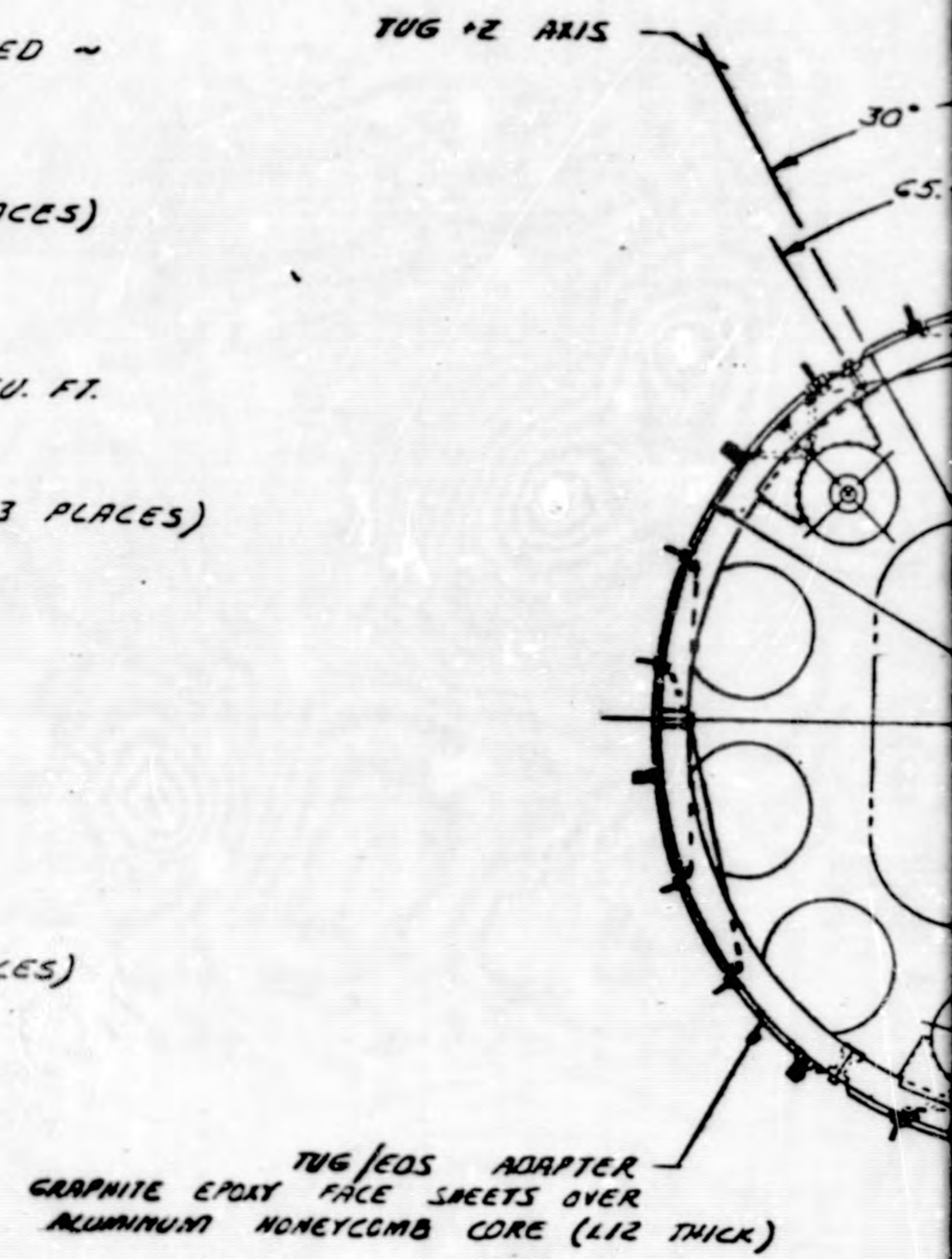


SECTION F - F SH 1

F-2

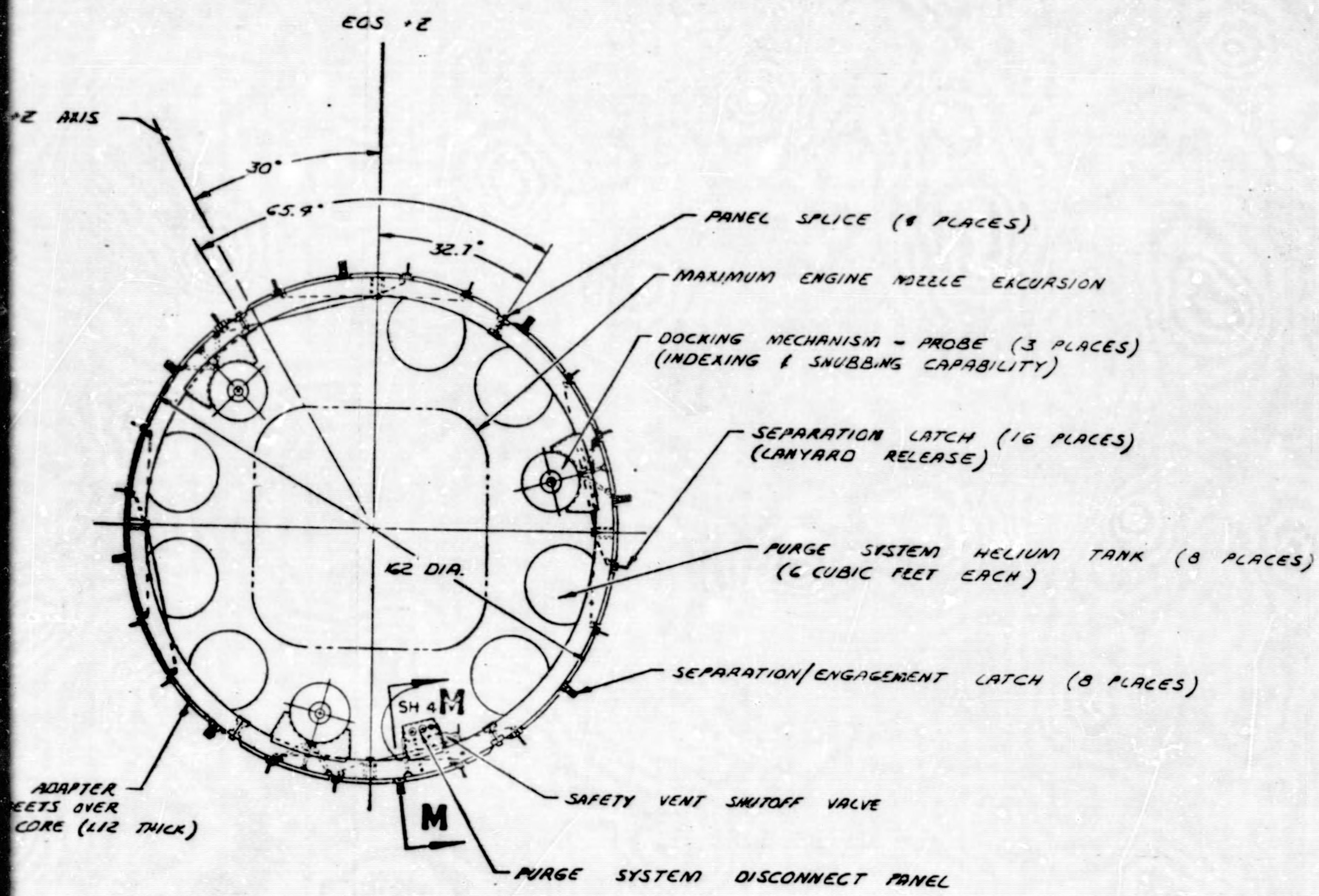


SECTION E - E SH 1



SECT

F-3



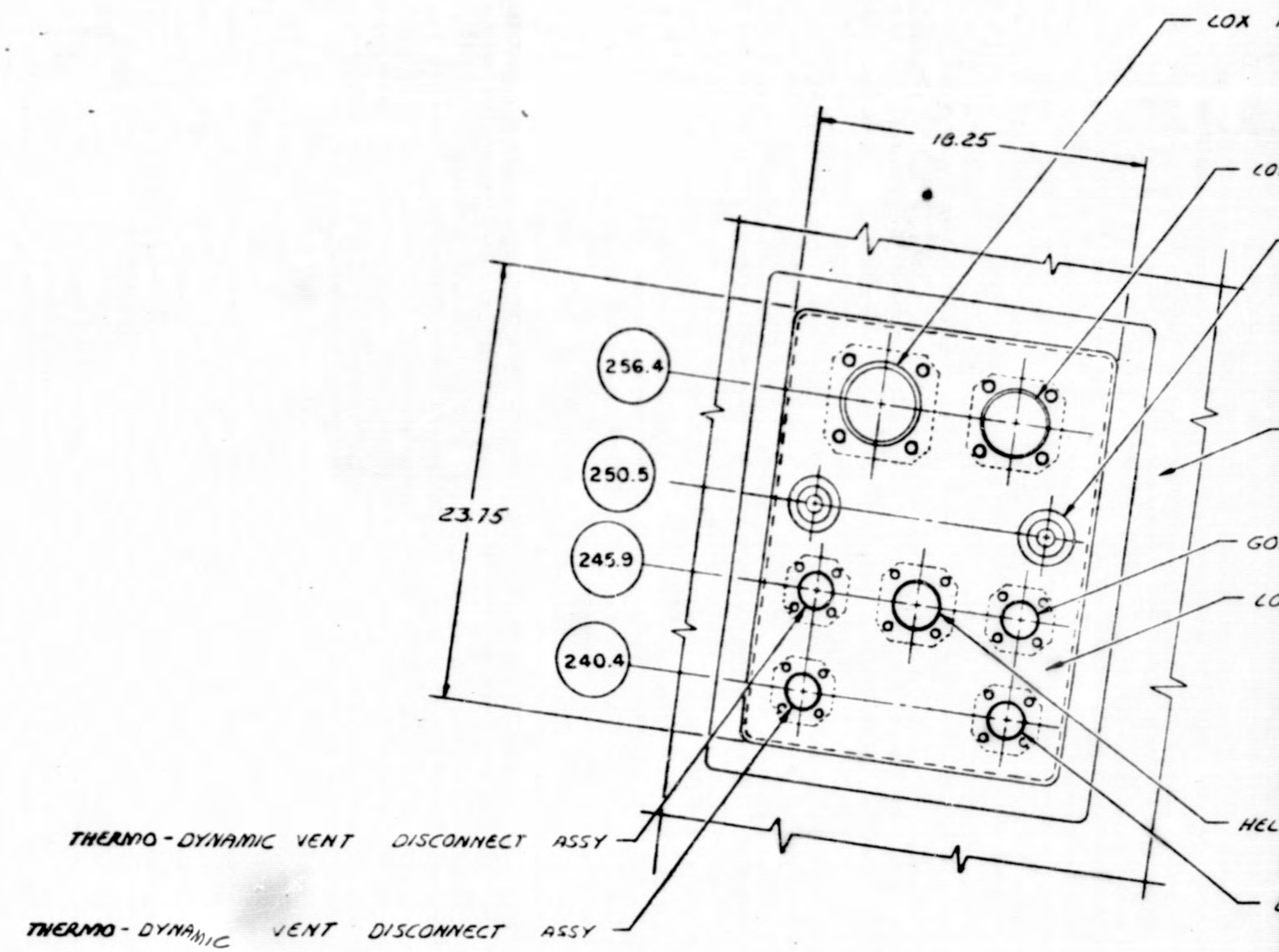
SECTION C – C SH 1

Figure 1.5-1
Inboard Profile Tug (Sheet 3 of 5)

F-4

I-13, I-14

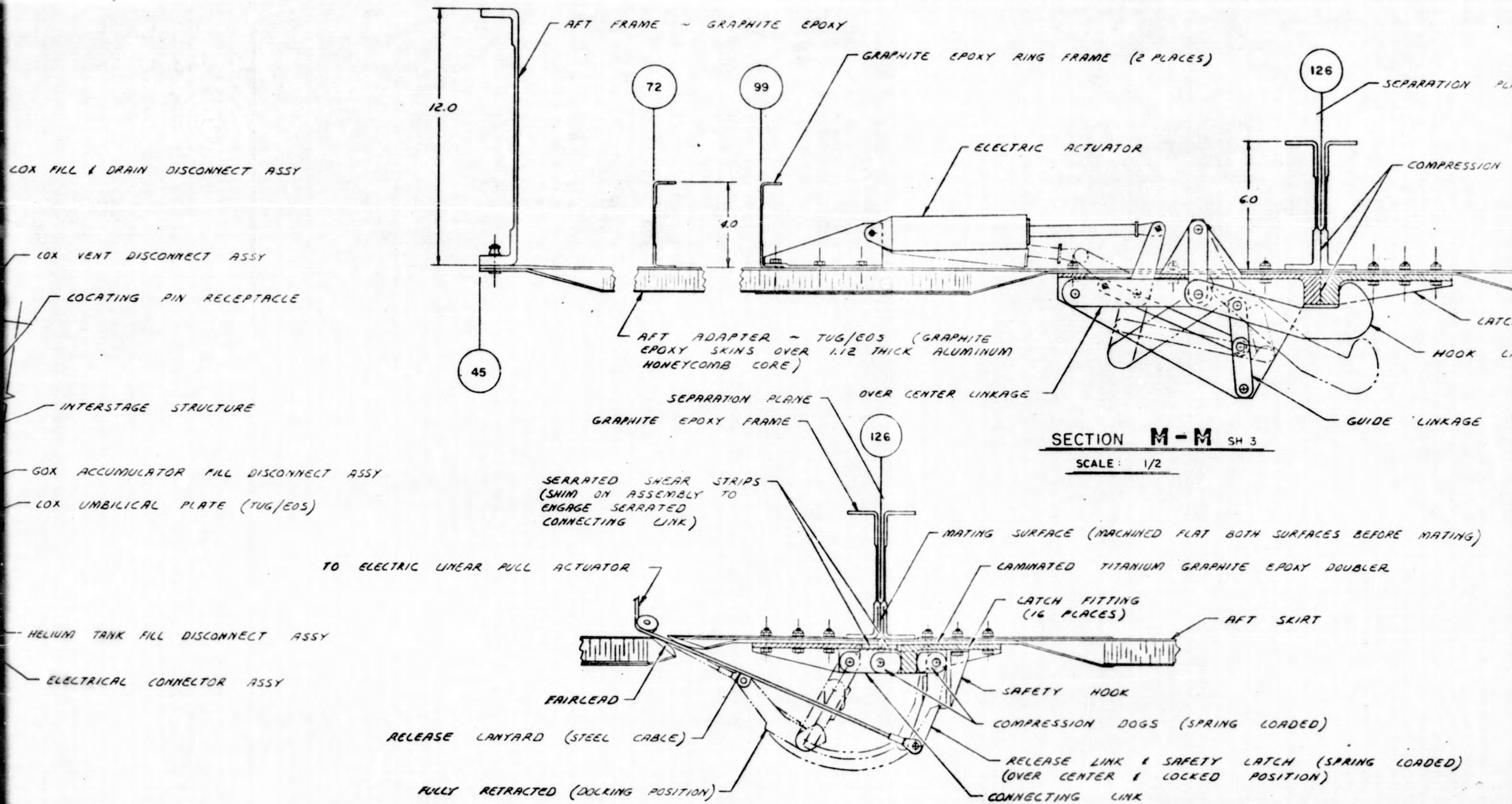
1/20	1-24-72 TUG	SPACE SYSTEMS NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
INBOARD PROFILE TUG			V7-923602 SH 3 OF 5



VIEW X - X SH 3

SCALE: 1/4

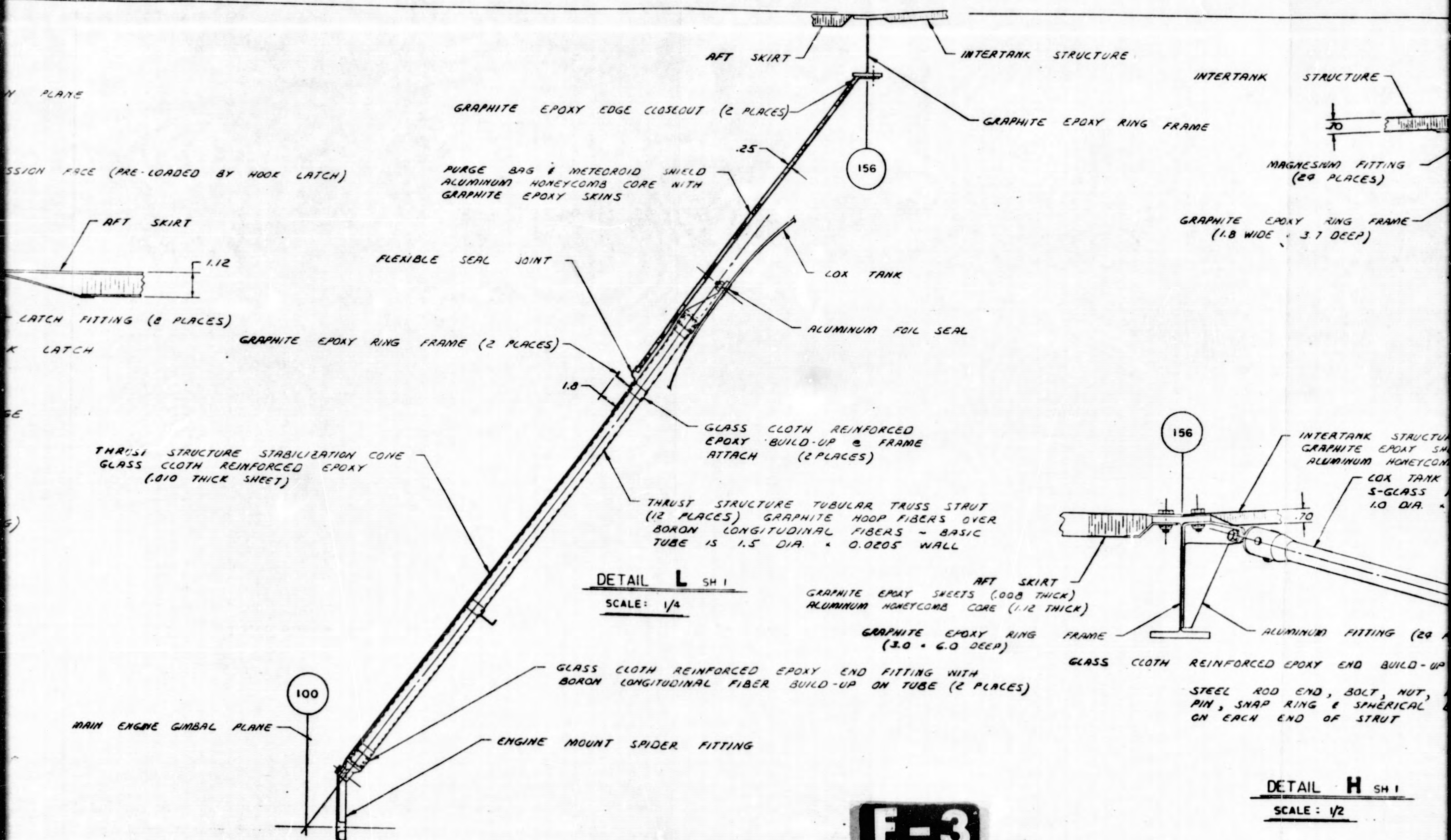
F-1



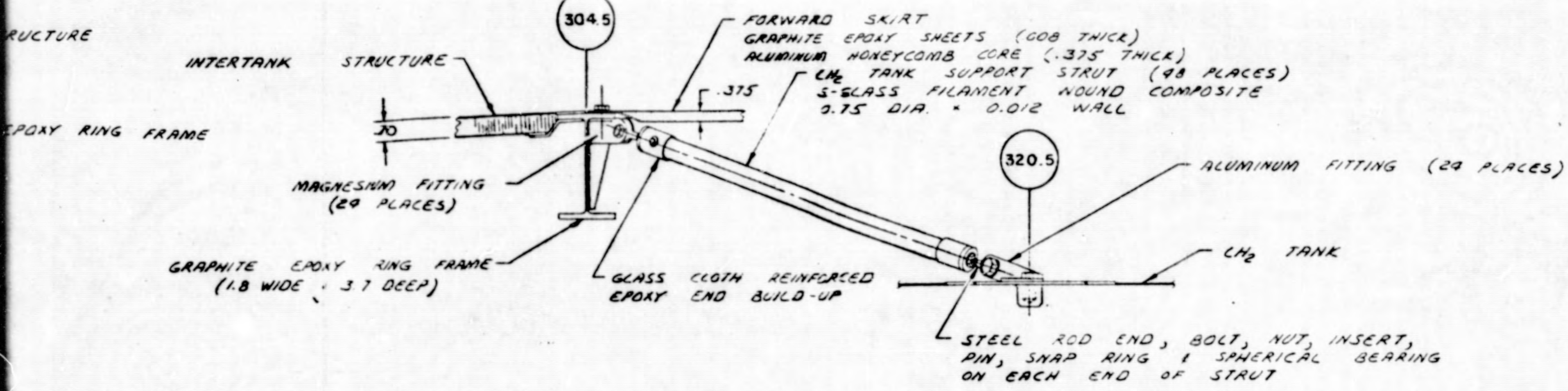
SECTION M-M SH 3
SCALE: 1/2

DETAIL K SH 1
SCALE: 1/2

F-2

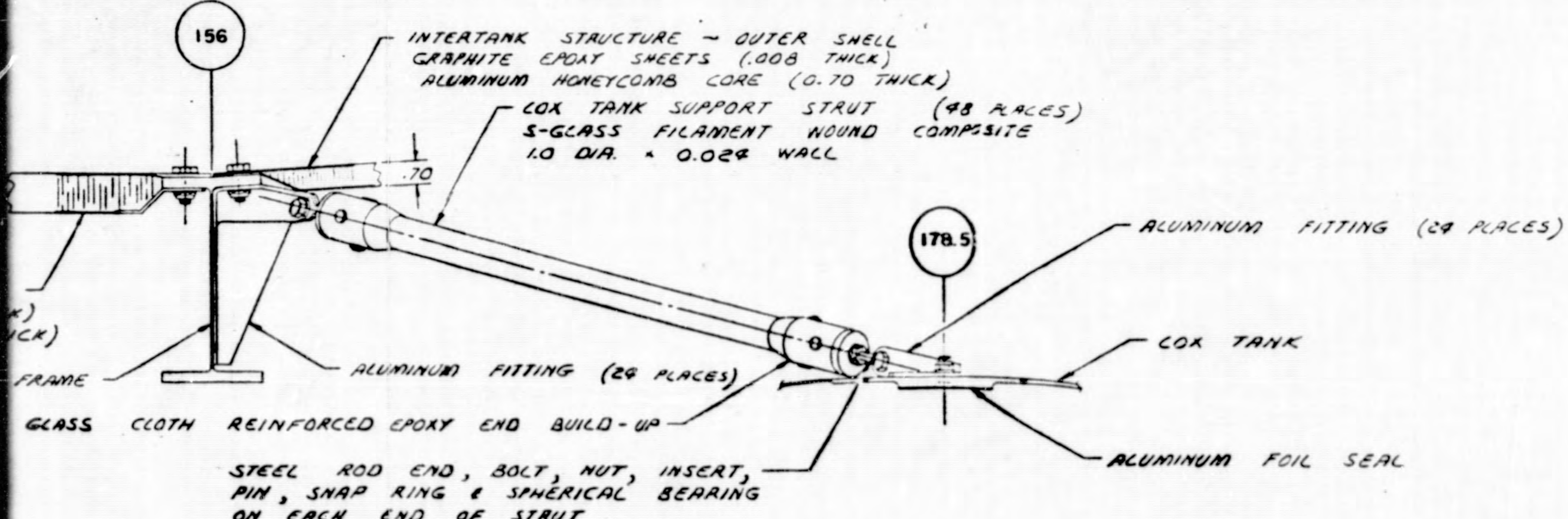

AFT SKIRT
INTERTANK STRUCTURE
GRAPHITE EPOXY EDGE CLOSEOUT (2 PLACES)
GRAPHITE EPOXY RING FRAME
156
.25
PURGE BAG & METEOROID SHIELD
ALUMINUM HONEYCOMB CORE WITH
GRAPHITE EPOXY SKINS
N PLANE
SSION FACE (PRE-LOADED BY HOOK LATCH)
AFT SKIRT
1.12
LATCH FITTING (8 PLACES)
K LATCH
FLEXIBLE SEAL JOINT
LOX TANK
ALUMINUM FOIL SEAL
GRAPHITE EPOXY RING FRAME (2 PLACES)
1.8
GLASS CLOTH REINFORCED
EPOXY BUILD-UP @ FRAME
ATTACH (2 PLACES)
THRUST STRUCTURE STABILIZATION CONE
GLASS CLOTH REINFORCED EPOXY
(.010 THICK SHEET)
THRUST STRUCTURE TUBULAR TRUSS STRUT
(12 PLACES) GRAPHITE HOOP FIBERS OVER
BORON LONGITUDINAL FIBERS - BASIC
TUBE IS 1.5 DIA. x 0.0205 WALL
DETAIL L SH 1
SCALE: 1/4
GLASS CLOTH REINFORCED EPOXY END FITTING WITH
BORON LONGITUDINAL FIBER BUILD-UP ON TUBE (2 PLACES)
100
MAIN ENGINE GIMBAL PLANE
ENGINE MOUNT SPIDER FITTING
INTERTANK STRUCTURE
.70
MAGNESIUM FITTING
(24 PLACES)
GRAPHITE EPOXY RING FRAME
(1.8 WIDE x 3.7 DEEP)
156
INTERTANK STRUCTU
GRAPHITE EPOXY SH
ALUMINUM HONEYCOM
LOX TANK
S-GLASS
1.0 DIA.
.70
AFT SKIRT
GRAPHITE EPOXY SHEETS (.008 THICK)
ALUMINUM HONEYCOMB CORE (1.12 THICK)
GRAPHITE EPOXY RING FRAME
(3.0 x 6.0 DEEP)
ALUMINUM FITTING (24
GLASS CLOTH REINFORCED EPOXY END BUILD-UP
STEEL ROD END, BOLT, NUT,
PIN, SNAP RING & SPHERICAL
ON EACH END OF STRUT
DETAIL H SH 1
SCALE: 1/2

F-3



DETAIL **J** SH 1

SCALE: 1/2



DETAIL **H** SH 1

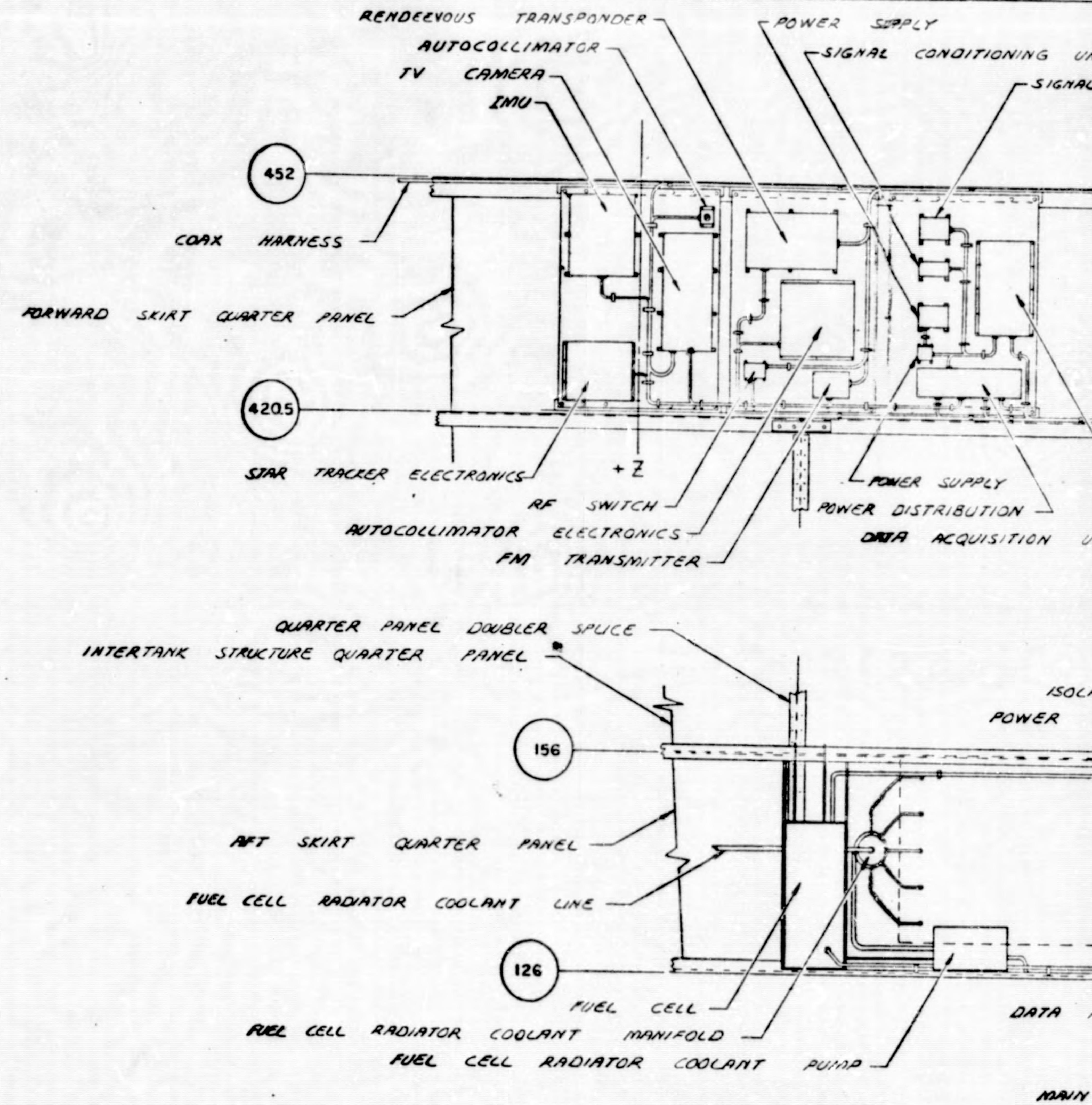
SCALE: 1/2

Figure 1.5-1
Inboard Profile Tug (Sheet 4 of 5)

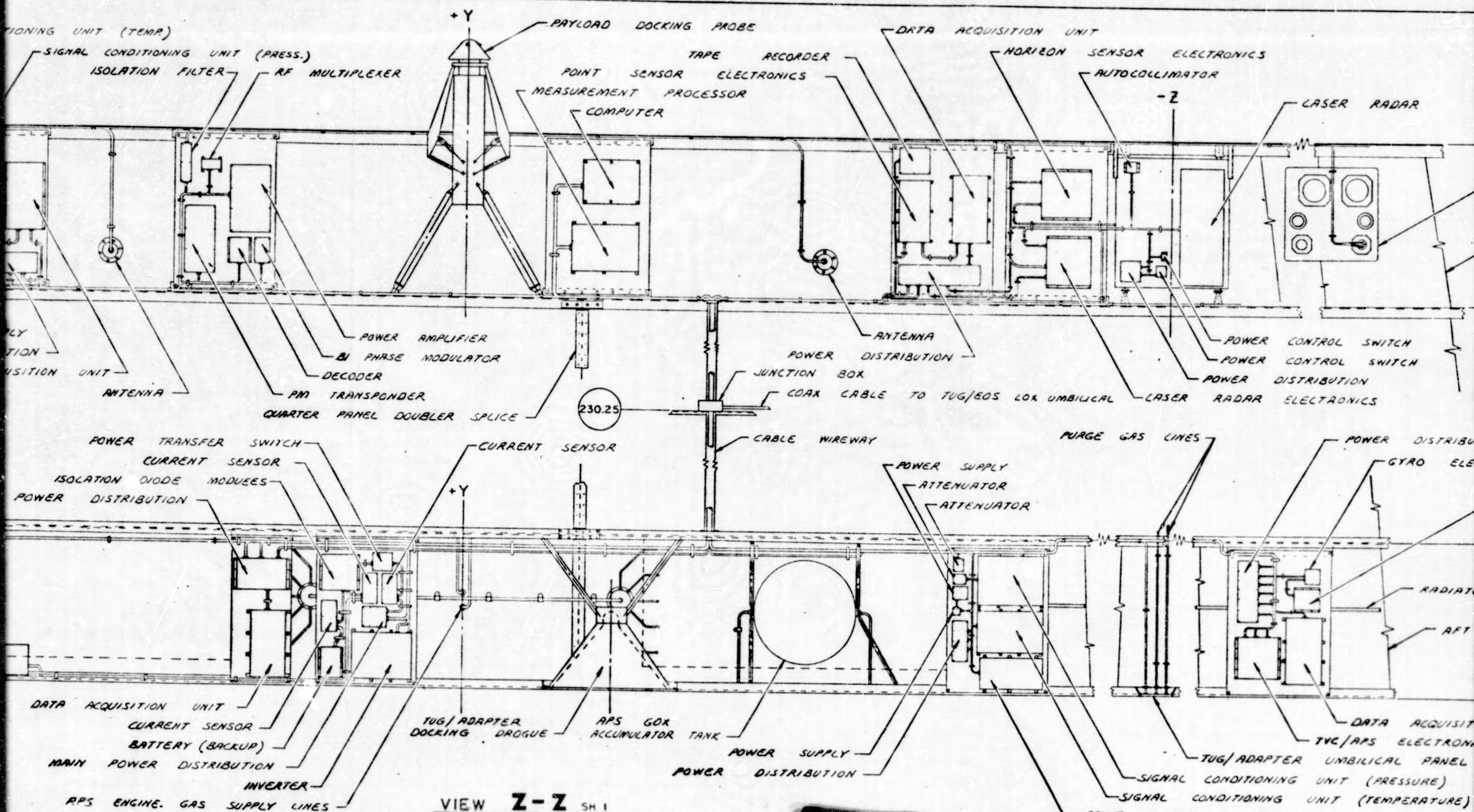
F-4

I-15, I-16

SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION	
INBOARD PROFILE TUG	V7-923602 SH 4 OF 5


RENDEZVOUS TRANSPONDER
AUTOCOLLIMATOR
TV CAMERA
IMU
POWER SUPPLY
SIGNAL CONDITIONING UNI
SIGNAL
452
COAX HARNESS
FORWARD SKIRT QUARTER PANEL
420.5
STAR TRACKER ELECTRONICS
+Z
RF SWITCH
AUTOCOLLIMATOR ELECTRONICS
FM TRANSMITTER
POWER SUPPLY
POWER DISTRIBUTION
DATA ACQUISITION UN
QUARTER PANEL DOUBLER SPLICE
INTERTANK STRUCTURE QUARTER PANEL
ISOLA
POWER
156
AFT SKIRT QUARTER PANEL
FUEL CELL RADIATOR COOLANT LINE
126
FUEL CELL
FUEL CELL RADIATOR COOLANT MANIFOLD
FUEL CELL RADIATOR COOLANT PUMP
DATA A
MAIN
APS

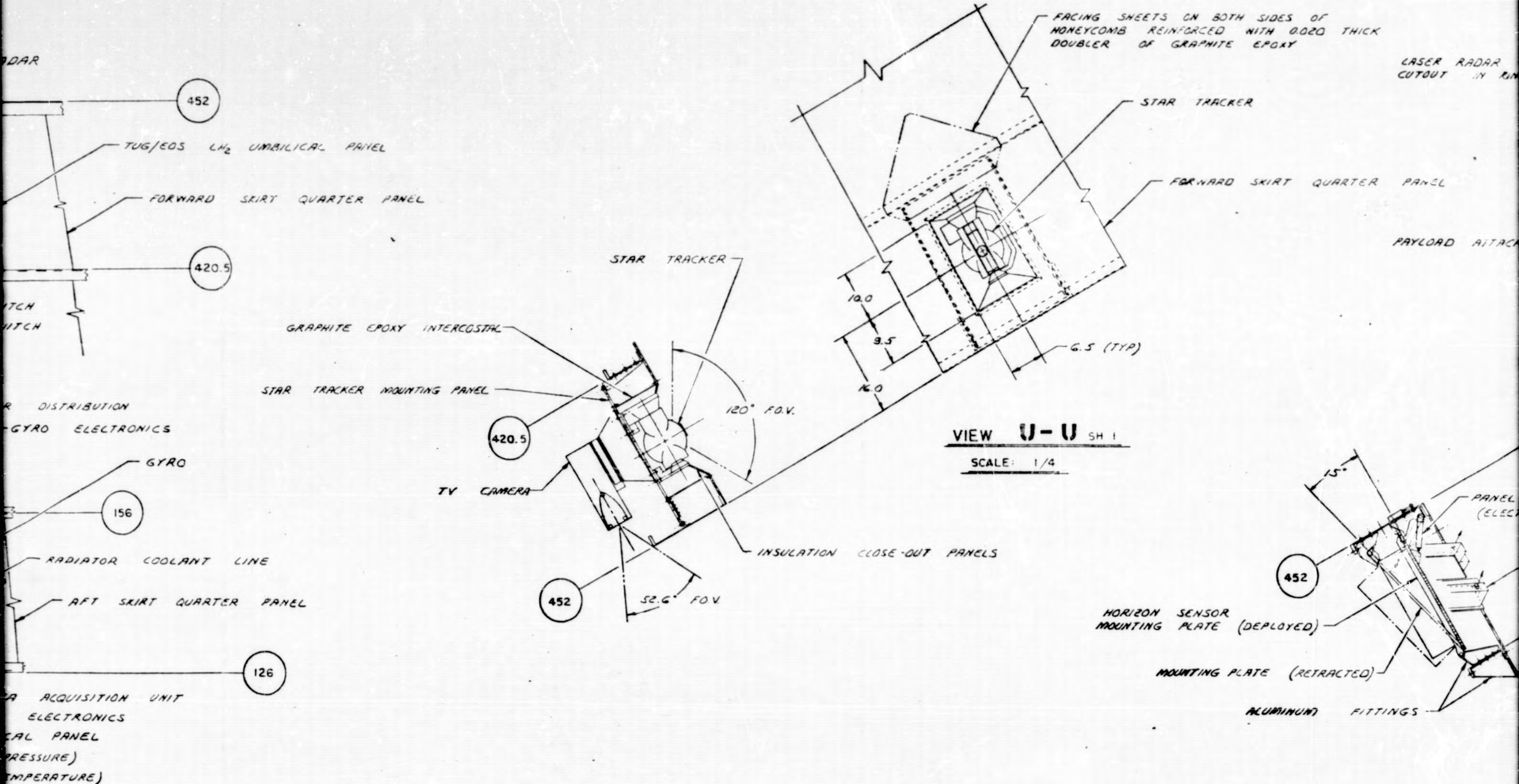
F-1



VIEW Z-Z SH 1

FORWARD & AFT SKIRT EQUIPMENT SHOWN ONLY IN FLAT PATTERN — SCALE: 1/8

F-2



ADAR
452
TUG/EOS LH_2 UMBILICAL PANEL
FORWARD SKIRT QUARTER PANEL
420.5
ITCH
ITCH
R DISTRIBUTION
-GYRO ELECTRONICS
GYRO
156
RADIATOR COOLANT LINE
AFT SKIRT QUARTER PANEL
126
A ACQUISITION UNIT
ELECTRONICS
CAL PANEL
PRESSURE)
MPERATURE)
STAR TRACKER
GRAPHITE EPOXY INTERCOSTAL
STAR TRACKER MOUNTING PANEL
420.5
120° F.O.V.
TV CAMERA
INSULATION CLOSE-OUT PANELS
452
52.6° F.O.V.
FACING SHEETS ON BOTH SIDES OF
HONEYCOMB REINFORCED WITH 0.020 THICK
DOUBLER OF GRAPHITE EPOXY
STAR TRACKER
FORWARD SKIRT QUARTER PANEL
10.0
9.5
16.0
6.5 (TYP)
VIEW U-U SH 1
SCALE: 1/4
LASER RADAR
CUTOUT IN RIN
PAYLOAD ATTAC
15°
PANEL
(ELEC
452
HORIZON SENSOR
MOUNTING PLATE (DEPLOYED)
MOUNTING PLATE (RETRACTED)
ALUMINUM FITTINGS

F-3

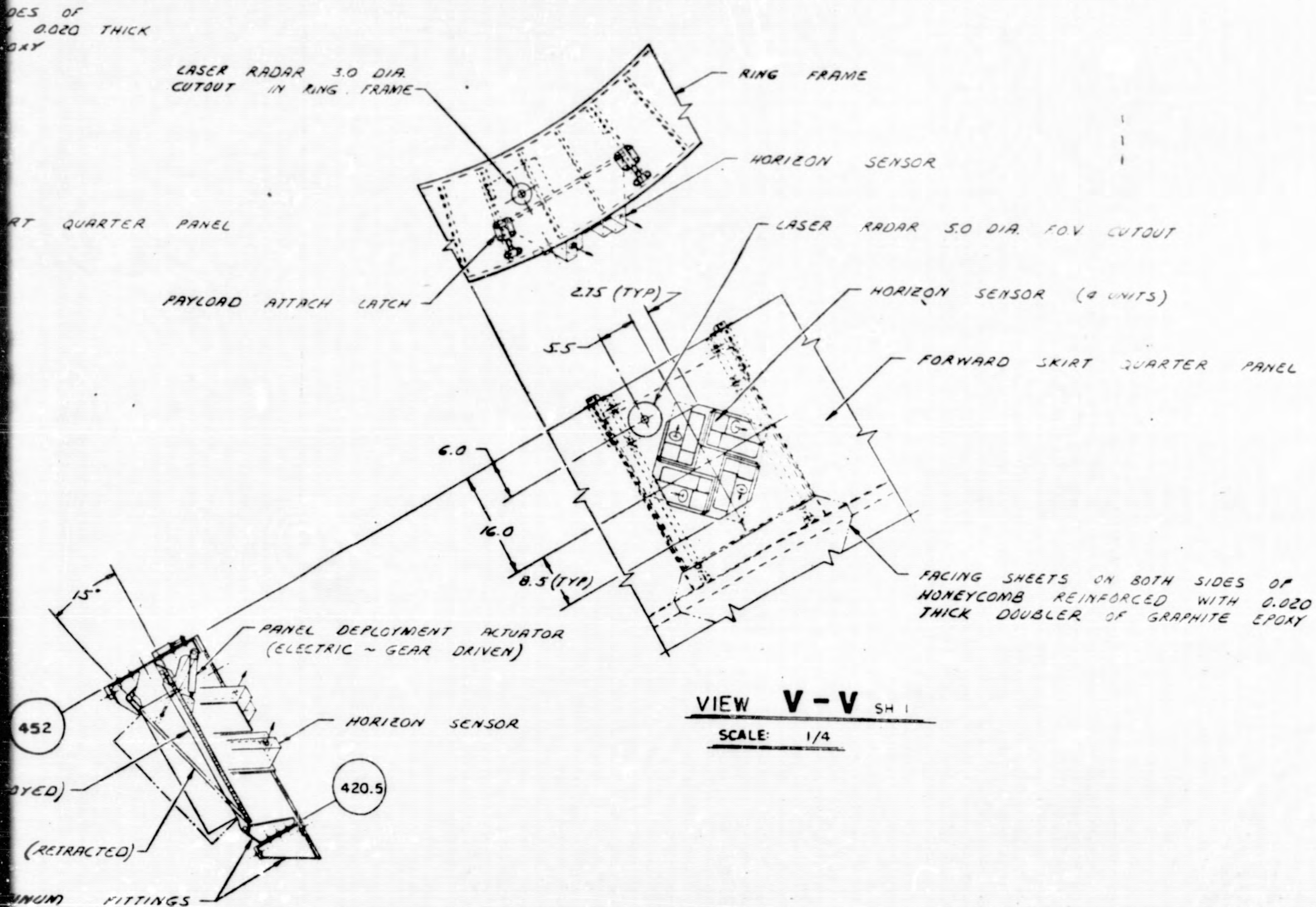


Figure 1.5-1
Inboard Profile Tug (Sheet 5 of 5)

F-3

I-17, I-18

SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA		
DATE 1-24-72	INBOARD PROFILE TUG	V7-923602 SH 5 OF 5

maintain the equipment within acceptable temperature limits during non-operating phases. These panels are attached to the shell with thermal isolation blocks which are mounted through brackets and fittings to the ring frames. The star tracker and horizon tracker panels are mounted 180° apart and are located on the Tug Z-axis. This axis is 30 degrees from the orbiter Z-axis, a condition which allowed both sensors a clear field of view with respect to the orbiter attach fittings. A cable wire way is provided at about the Tug y-y axis and is supported from the stability frames. In this location it is centrally located between the 8 avionics panels. In various positions within the shell structure, clips and brackets are provided on the inboard caps of the stability ring frames to attach the various lines and controls associated with the main propulsion system, auxiliary propulsion system and pressurization and vent subsystems.

A purge bag/meteoroid shield is provided at the forward end of the forward skirt. This shield is a spherically contoured diaphragm of rubber impregnated glass cloth. It is attached near the inboard cap of the 6-inch frame, aft of the interface. The diaphragm serves as both the container for the purge gas, as well as a meteoroid bumper. All of the avionics panels are forward of this barrier and consequently are accessible from the forward end of the Tug.

The forward frame provides for the support of a payload docking system. Consequently, docking and latching provisions are incorporated on this ring frame. The payload is attached to the Tug through 24 electrically actuated latches. Acquisition and docking is accomplished by the use of 3 Apollo type probes. These probes would be simplified versions of the Apollo-LEM docking probes since many of the complex operational requirements associated with the Apollo missions are not requirements for the Tug. Three off-center probes were selected since a single centrally located probe could not be installed on the Tug without greatly increasing the length of the vehicle. The probes are supported between the forward two frames and therefore extend forward of the interface plane to accomplish initial docking with the payload.

The final hard latching is accomplished by the 24 latching fingers spaced around the periphery of the forward frame. The latches are located at 15 degrees interval to assure uniform distribution of loading. The latching fingers are translated by gearing and are electrically activated. Additional redundancy may be obtained by the use of dual, parallel drive motors or by the use of an override pyrotechnic device system.

The Tug/EOS LH_2 umbilical panel is located on the forward skirt between the two forward frames. The panel is located such that when in the orbiter, the panel faces toward the lower belly corner of the orbiter. The LH_2 fill and drain line passes through a sealed cutout in the purge bag and interfaces with the panel. In addition, the panel accommodates pressurization and vent lines, as well as electrical connections. The various plumbing lines associated with the main propulsion, APS, and pressurization and vent system, are covered with insulation to reduce heat leaks to the main tanks and minimize boil-off.

The LH_2 tank is 168 inches in diameter with hemispherical bulkhead. A short cylindrical section (37.5 inches) separates the two bulkheads. The bulkheads are made up of six preformed gore sections and a central circular section which are all butt-welded together. These gore sections are 0.080 inch thick 2014 T651 aluminum sheet chem-milled to a thickness of 0.020 inches between the weld bands. The forward bulkhead has a 24-inch diameter opening access door located in the center of the circular section. The aft bulkhead has an identical circular section but contains a dollar weld section rather than an access door.

The short cylindrical section is made up of three 2014-T651 aluminum sheets (0.10 inch thick) which are butt welded into the cylindrical shape. The cylinder is then chem-milled to a thickness of 0.045 inch between the weld lands. At the forward end, the cylinder is butt-welded directly to the forward bulkhead. The aft end of the cylinder, however, is butt-welded to a heavier ring segment which is then butt-welded to the aft bulkhead. This heavier ring provides the thickness necessary to attach the strut truss support fittings. It also serves to adequately distribute the loads induced into the tank from the tubular strut truss support which suspends the tank within the outer shell structure. Various bosses are welded to the tank to accommodate the tank penetrations required for feed, fill and drain, pressurization, vent and electrical lines. These penetrations have been grouped together to minimize the number of bosses within a gore section. The forward bulkhead access door contains the fill and drain penetration and also accommodates the vent deflector, which is an integral part of the door.

The APS LH_2 auxiliary tank is mounted inside the main LH_2 tank on the aft bulkhead. Its' centerline is located 3 inches from the center of the tank and is supported 1 inch from the bulkhead. It is supported on 6 lugs which are welded to the gore section weld lands. These lugs are separated 1.375 inches so as not to interfere with the inspection of the gore section welds. Other clips and lugs are similarly welded within the tank on the weld lands to provide attach points for the settling baffle support struts and the point sensor stillwell support guy wires.

The LH_2 tank is completely covered with multi-layer insulation. The insulation can be divided into four sections: the forward access door segment, the aft dollar weld segment, the aft bulkhead segment (up to the support struts) and the forward bulkhead and cylindrical section. The forward and aft bulkhead and cylindrical section are made up of 24 pie-shaped slices which are staggered when installed on the tank. The forward section covers the forward bulkhead, the cylindrical section, and contains the penetrations for the outer shell/tank support struts. The aft section covers only the aft bulkhead. The dollar weld section contains the penetrations for the feed, vent, APS fill and feed and the electrical lines, and is configured as a circular section. The MLI is spaced from the tank wall, a distance of one inch by foam filled honeycomb pads which are bonded directly to the tank wall. An aluminum wire mesh internal support membrane is then fastened to the pads through threaded inserts in the pads. This wire mesh is stretch formed to the correct contour

of the tank. The MLI layers are then attached with fiberglass epoxy tube posts which are installed to the pads through threaded inserts (not the same as used for the wire mesh). An external tension membrane of Nomex mesh is then applied over the entire surface to prevent ballooning of the insulation. This membrane is applied in quadrants for the forward and aft sections, and a band for the area where the struts penetrate the MLI.

Twenty-four aluminum fittings are attached to the LH_2 tank outer wall to accommodate the support struts. The S-glass filament wound composite tubular struts suspend the tank within the shell. There are a total of 48 struts in the support truss. The ends of the tubes are reinforced with glass cloth epoxy build-up layers and are attached to the tank fitting through steel rod ends which are adjustable. The rod ends are mounted through spherical bearings to the fittings. The tubes are 0.75 inch in diameter with a 0.012 inch wall. At the outer shell they are attached with identical mounting at the tank end to magnesium fittings which are attached to a graphite epoxy composite "I" beam frame at the forward end of the intertank structure. The forward skirt is attached at this same frame with "Hi Lok" fasteners. The "I" beam frame thus functions as the field/manufacturing joint interface between the forward skirt and the intertank shell.

The intertank shell structure is a truncated cone of 180 inches in diameter at the forward end and 162 inches in diameter at the aft end. It is 148.5 inches long. The construction is identical to that of the forward skirt (aluminum honeycomb core with graphite epoxy skins). The shell is 0.70 inches thick and in addition to the "I" beam frame at the forward end, incorporates three channel section ring frames of graphite epoxy which are equally spaced and measure 0.5 inch by 1.75 inches deep. As in the forward skirt, these frames are primarily for stability and in addition provide for mounting of clips and brackets to support the various propulsion system plumbing lines and controls, and the electrical cable wire way and junction box.

Several penetrations of this shell are accommodated for the Shuttle/Tug LOX umbilical, purge venting and an access door to allow servicing of the intertank area. The access door is located between the two aft ring frames and is a 30-inch square opening. Two six-inch diameter opening vent valves are located 180 degrees apart in the forward area of the shell. These valves vent the purged intertank area to allow rapid evacuation of the MLI during orbiter ascent. The LOX umbilical panel is located between the two forward stability frames and is oriented such that it faces the belly or lower portion of the orbiter when the Tug is installed in the cargo bay.

At the aft end of the shell, located 90 degrees apart are four 30 x 48-inch panels of dynaflex R-F 2400 insulation covered with 0.002 inch thick stainless steel. These panels are to accommodate the forward firing APS engine plume impingement in this area. The panels are attached to the outer surface of the shell. As in the forward skirt the intertank shell is made in quarter panels and the splicing of these panels is identical to that described for the forward skirt. The intertank structural shell is attached at its aft end to the aft skirt, with "Hi Lok" fasteners. It is attached to a graphite composite "I" beam which is part of the aft skirt.

The aft skirt is a cylindrical shell assembly 162 inches in diameter and 30 inches long. It is identical in construction to the other outer shell assemblies but is 1.12 inches thick. It is made up of quarter panels of graphite epoxy skins over aluminum honeycomb core. There are no intermediate or stability frames in this skirt, only the forward "I" beam frame and the aft channel frame. The aft frame mates with an identical counterpart on the aft adapter, which supports the Tug within the orbiter cargo bay. This skirt also accommodates the mounting of three docking drogues for the Tug/Shuttle adapter docking. These drogues are supported by an intercostal to the aft frame and stabilized by a tubular strut truss between the forward and aft frame. Two APS gas accumulator tanks and an engine purge system helium supply tank are mounted on the aft skirt with simple girth attached "VEE" strut trusses with a torsional stability strut. These tubular struts are also attached to the frames. Four avionics panels, identical to those in the forward skirt, are also mounted between the frames in the aft skirt. None of these panels require louvers but all have heaters to maintain temperatures within acceptable limits during non-operating phases of the missions. The avionics equipment located in the aft skirt is that which is propellant or propulsion oriented. The aft frame interface is also the interface for the purge system disconnect panel between the shuttle mounted adapter and the Tug. A fuel cell and associated coolant pump and controls are mounted on intercostals between the two frames. The coolant is manifolded to four radiator panels located on the aft skirt outer skin 90 degrees apart. These radiator panels are connected in series to maintain a constant temperature to the fuel cell regardless of the orientation of the Tug with respect to the sun.

Two five-engine APS modules are mounted on the outer surface of the aft skirt between the radiator panels. The two quints are 180 degrees apart and between them are located two dual engine modules. The quints are located on the Tug Z-Z axis while the duads are located on the Y-Y axis. The duads have forward and aft firing engines. The quints are composed of conventional type quad firing engine patterns with the addition of a single engine firing radially outward. Various clips and support brackets are provided by the aft skirt for attaching and supporting plumbing lines, cables and an electrical cable wire way and junction box. The aft skirt contains 24 latch receptacle fittings at the aft end to mate the Tug with the Shuttle mounted adapter latching system. Twenty-four aluminum fittings at 15 degrees interval, are attached to the forward frame to provide attachment of the LOX tank support struts.

Forty-eight tubular struts attach the LOX tank to the outer shell. These struts are S-glass filament wound tubes which are identical to those described for the LH_2 tank except for the larger diameter (1 inch) and wall thickness (0.024 inches). The ends of the struts contain adjustable rod ends and spherical bearings. At the LOX tank the struts are attached to aluminum fittings at the girth of the tank.

The LOX tank is an oblate spheroid 150 inches in diameter and 105 inches high and is manufactured from 2014-T651 aluminum. The tank is made up of the two bulkheads and a girth ring which are all butt-welded together. Each bulkhead is composed of six preformed gore sections. There is a large diameter (34 inches) dollar weld on the aft bulkhead and a small diameter (14 inch)

dollar weld on the forward bulkhead. The 6-inch girth ring is 0.30 inches thick. Adjacent to this on the forward bulkhead the tank thickness tapers from 0.125 to 0.040 inches which is the basic thickness for this bulkhead. The aft bulkhead is basically 0.030 inches thick but tapers to 0.125 inches thick in the area where the thrust structure is attached.

The penetrations to the tank are grouped into one area on the aft bulkhead and all bosses are located on a circular thickened section in one gore section. In the forward bulkhead, the fill and drain inlet is a similarly handled penetration. An access door is located off center in the forward bulkhead and 30 degrees from the Tug + Z axis. This door offers a 20-inch opening into the LOX tank.

An APS LOX auxiliary tank is located on the aft bulkhead within the main tank. This tank is similar to that in the LH_2 tank. It is put into the tank in halves through the opening in the forward bulkhead. It is attached to the aft bulkhead by lugs which are welded to the gore section weld lands and the tank is mounted one inch from the inside wall of the bulkhead. Other tabs welded to the weld lands and thickened sections allow attachment and support points for the settling baffles (three baffles) and the sensor probe stillwell guide cables.

The LOX tank, like the LH_2 tank, is completely covered with MLI. This insulation is applied in 24 gore segments in a staggered configuration. The MLI is divided into the forward bulkhead section, the aft bulkhead section, and a dollar section for the forward and aft apex. The forward section extends beyond the tank support struts and consequently contains the penetrations and seals for these struts. These MLI panels are applied in exactly the same manner as those described for the LH_2 tank (foam filled honeycomb hardspots, aluminum wire mesh, 0.50 inch thick MLI and an external tension membrane). In the area of the access door, special provisions have been made to remove the insulation and screen mesh to permit opening of the tank. Twelve penetrations of the aft bulkhead insulation accommodate the thrust structure struts. The one-inch gap between the tank wall and the insulation serves as the purge gas annulus for the insulation and is purged through an aluminum tube manifold at the girth of the tank. This manifold distributes the gas under the MLI. This concept is identical to that used on the LH_2 tank.

The thrust structure is attached directly to the aft bulkhead on the LOX tank. The primary load carrying structure is composed of the 12 tubular struts which attach the engine gimbal block mount to the LOX tank. These struts are 1.5 inches in diameter and are made up of a combination of boron epoxy longitudinal fibers and graphite epoxy circumferential fibers. The struts are 60 inches long with built-up ends to accommodate the end fittings for attachments. A glass cloth reinforced epoxy bathtub type fitting on the aft end attaches to the engine gimbal block mount aluminum spider fitting with "Hi Lok" fasteners. At the forward end, the tub terminates in another glass cloth reinforced epoxy bathtub type fitting which attaches directly to the LOX tank, at a tangency point. The thrust structure tubes are flattened on two sides to accommodate the attachment of ring frames and a truncated torsional stability cone.

The stability cone is a glass cloth reinforced epoxy skirt 0.010 inch thick. It is attached to the flat areas on the struts. Two graphite epoxy channel ring frames are attached to the thrust structure with blind pull rivets. One frame is attached to the inner portion of the cone and one to the outer surface. The tubes are built up in the areas where these frames are attached. Brackets and tabs on the frames and the cone structure provide for mounting of APS equipment. Two APS conditioning units are attached to the thrust structure, as well as controls, plumbing and electrical lines. Two of the struts provide attach fittings for the main engine gimbal actuators. The thrust structure also provides aft meteoroid protection for this vehicle, as well as serving as the aft purge bag. To continue the meteoroid protection/purge bag, an additional truncated cone of graphite epoxy skins (0.008 inches thick) over aluminum honeycomb core (0.25 inches thick) is provided between the thrust structure and the outer shell. This cone is attached with "Hi Lok" fasteners to the aft skirt forward frame and is joined to the thrust structure at the forward channel frame through a flexible seal, which does not allow this structure to carry thrust loads.

The feed lines to the main engine are covered with multi-layer insulation as are the fill and drain lines for the main tanks. This insulation is applied directly to the lines and does not provide a purge annulus. The vent lines and APS feed lines are also insulated. The APS accumulator tanks mounted on the aft skirt are also covered with MLI. This insulation is applied directly to the tank. A thermal coating is applied to the aft facing surface of this insulation. All of the insulation on the plumbing lines and on the accumulator tanks is held in place with zippered tension membranes similar to that on the main tanks.

The Tug purge bag consists of the outer shell structure, the forward spherically shaped membrane, the thrust structure, and the small closeout between the thrust structure and the outer shell. To insure proper sealing of this structure/purge bag and reduce leakage of the purge gas, the entire inner surface of the purge bag is covered with a sealant coating of RTC epoxy resin. The large area around and between the main tanks is purged by the same system that purges the main tanks and the insulation. Gas diffusers are located in the intertank area for this purpose. The entire outer surface of the Tug vehicle is covered with a thermal control coating of zinc oxide in RTV methyl silicone. This coating is applied to the outer surface of the outer shell, the forward meteoroid shield/purge bag outer surface, the outer surface of the thrust structure and the inner surface of the aft skirt.

The Tug is attached inside the orbiter by a cylindrical shell shuttle mounted adapter. The Tug is upside down when installed in the orbiter cargo bay, and is supported from the forward bulkhead. This adapter is 162 inches in diameter to mate with the aft skirt and is 81 inches long. It is assembled from four panels which are made up of aluminum honeycomb core with graphite epoxy composite face sheets. The honeycomb core is 1.12 inches thick and the facing sheets are 0.058 inches thick in the major load carrying areas and 0.008 inches thick in the lower stressed areas.

The shell is reinforced by four equally-spaced channel section, graphite epoxy ring frames. The aft frame is 12 inches deep and the intermediate frames are 4 inches deep each. The shell is comprised of two panels which subtend an angle of 65.4 degrees each and two which subtend an angle of 114.6 degrees each. Titanium is used for fittings and doublers for assembling the cylinder.

The adapter provides the main structural attachment for the Tug to the orbiter cargo bay and contains Tug deployment and docking provisions. Once the Tug is separated from the orbiter, this adapter structure remains with the orbiter. The launch load, are transmitted to the adapter structure through two interconnected, load equalizing hydraulic struts which interface with the adapter at two trunnions located on the aft end of the adapter and slightly aft of the aft frame. Pitch loads are introduced tangentially through these fittings and yaw loads are introduced tangentially through a fitting at the bottom or belly portion of the cylinder.

The adapter houses the helium purge system for the main tanks and insulation. Eight 6-cubic feet tanks are mounted near the aft end of the adapter for the helium storage. Various controls and plumbing are accommodated, as well as a disconnect panel which interfaces with the Tug through the aft skirt disconnect. For deployment of the Tug vehicle out of the orbiter cargo bay, a shaft located near the upper forward corner of the orbiter cargo bay is inserted into receiver fittings attached to the adapter. The adapter and Tug combination is then rotated out of the cargo bay about this shaft.

At the Tug/adapter interface, 24 latches are located at 15 degrees interval around the perimeter of the adapter. Sixteen of these latches are folded away after initial deployment, leaving only the 8 independently actuated latches which are used for redocking and reattachment of the Tug for re-entry, when the Tug is empty and the loads are consequently lower. The sixteen latches are compression dogs with an overcenter link which are initially locked on the ground and are released in space with a pull cable system. The eight remateable latches are hook latch type with overcenter linkage and are electrically driven.

Once the Tug mission is completed, docking of the Tug to the adapter is accomplished by three probes in the adapter. When extended, the probe arms are free to deflect laterally under slight pressure. The probes engage drogues in the aft skirt and are then retracted to draw the Tug to the adapter. Spherical bearings and an axial alignment system within the probes insure that the Tug is aligned correctly with the adapter for engagement of the eight hook latches. When the Tug/adapter interface has been made, the eight latches are actuated and the final attachment is accomplished. The Tug/adapter combination is then rotated back into the orbiter cargo bay, the trunnion attachment fittings are engaged and the deployment shaft removed from the fittings on the adapter.

HARDWARE TREE

The Hardware Tree shown in Figure 1.5.2 is a summary of all the components comprising the four basic systems of the Tug, with the exception of lines,

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SPACE TUG

STRUCTURE (S)

LH$_2$ TANK (SHT)

1. BULKHEADS
2. SIDEWALLS
3. ACCESS DOOR
4. TANK SUPPORTS
5. BAFFLES
6. START TANK SUPPORTS

LOX TANK (SOT)

1. BULKHEADS
2. ACCESS DOOR
3. TANK SUPPORTS
4. BAFFLES
5. START TANK SUPPORTS

THRUST STRUCTURE (ST)

1. THRUST BLOCK SUPPORTS
2. THRUST BLOCK
3. SKIN PANELS
4. FRAMES
5. SUPPORT STRUTS
6. SYSTEM SUPPORT

FORWARD SKIRT (SFS)

1. FRAMES
2. SIDEWALL
3. SHUTTLE ORBITER ATTACH FITTINGS
4. SYSTEM SUPPORT

INTERTANK (SI)

1. FRAMES
2. SIDEWALL
3. ACCESS DOOR
4. SYSTEM SUPPORT

AFT SKIRT (SAS)

1. FRAMES
2. SIDEWALL
3. SYSTEM SUPPORT

DOCKING (SD)

1. LATCHING MECHANISM
2. DOCKING PROBE (PAYLOAD)

AVIONICS (A)

DATA MANAGEMENT (ADM)

1. DIGITAL COMPUTER
2. DATA ACQUISITION UNIT
3. MEASUREMENT PROCESSOR UNIT
4. INTERFACE UNIT
5. TAPE RECORDER

GUIDANCE, NAVIGATION & CONTROL (AGN)

1. INERTIAL MEASURING UNIT
2. STAR TRACKER
3. HORIZON SENSOR
4. AUTO COLLIMATOR
5. ENGINE CONTROL ASSEMBLY
6. BACKUP STABILIZATION ASSEMBLY

COMMUNICATIONS & INSTRUMENTATION (ACI)

1. TRANSPONDER - PM
2. TRANSMITTER - FM
3. POWER AMPLIFIER
4. BI-PHASE MODULATOR
5. DECODER
6. OMNI ANTENNA
7. HYBRID JUNCTION
8. DIRECTIONAL COUPLER
9. ISOLATION FILTER
10. R.F. SWITCH
11. R.F. MULTIPLEXER
12. STRAIN GAGE PRESSURE TRANSDUCER
13. POTENTIOMETRIC PRESSURE/POSITION TRANSDUCER
14. TEMPERATURE TRANSDUCER
15. CURRENT SENSOR
16. FLOW METER
17. LIQUID LEVEL POINT SENSOR
18. STRAIN GAGE SENSOR AMPLIFIER
19. TEMPERATURE SENSOR AMPLIFIER
20. CURRENT SENSOR ELECTRONICS
21. LIQUID LEVEL POINT SENSOR ELECTRONICS
22. ATTENUATOR
23. 5 VDC REGULATED POWER SUPPLY
24. ± 28 VDC POWER SUPPLY

ELEC POWER GENERATION, CONTROL & DIST (AEP)

1. FUEL CELL
   1. REACTOR STACK
   2. ISOLATION VALVE
   3. PUMP
   4. REGULATORS
   5. CONTROL VALVES
   6. CONDENSER
   7. WATER TANK
   8. CHECK VALVES
   9. VENT VALVES
   10. SCRUBBERS
   11. EJECTORS
   12. GAS SEPARATOR
2. BACKUP BATTERY
3. POWER TRANSFER SWITCH
4. POWER BUS MODULE
5. RESISTOR & DIODE MODULE
6. POWER CONTROL SWITCH

RENDEZVOUS AND DOCKING (ARD)

1. LAZER RADAR
2. TELEVISION CAMERA
3. RENDEZVOUS TRANSPONDER
4. OMNI ANTENNA
5. POWER DIVIDER

PROPULSION (P)

MAIN PROPULSION (PMP)

1. LOW SPEED INDUCER
2. TURBOPUMP
3. ISOLATION VALVE
4. PREBURNER
5. CONTROL VALVES
6. REGULATOR
7. RECEIVER
8. CHECK VALVE
9. HEAT EXCHANGER
10. ENGINE

PROPELLANT FEED FILL & DRAIN (PPF)

1. THERMAL ISOLATION SEGMENT
2. PREVALVES
3. FILL & DRAIN VALVES
4. DISCONNECTS
5. J-T EXPANSION VALVES
6. REGULATORS
7. CONTROL VALVES

PROPELLANT MANAGEMENT (PPM)

1. LEVEL SENSORS
2. SENSOR SUPPORT MAST
3. CAP PROBE

PROPELLANT ORIENTATION (PPO)

1. AUXILIARY PROPELLANT TANKS
2. PROPELLANT LEVEL SENSORS

THRUST VECTOR CONTROL (PTC)

1. PUMP MOTOR ACTUATOR
2. RESERVOIR/ACCUMULATOR ACTUATOR

PRESSURIZATION (PPR)

1. CHECK VALVE
2. REGULATORS
3. DIFFUSERS
4. DISCONNECTS
5. CONTROL VALVES

SAFING & VENTING (PSV)

1. RELIEF VALVES
2. VENT VALVES
3. PRESSURE SWITCHES
4. SELECTOR VALVES
5. CONTROL VALVES
6. DISCONNECTS
7. CHECK VALVES
8. NON-PROPULSIVE NOZZLES

REACTION CONTROL (PRC)

1. BURST DISCS
2. SQUIB VALVES
3. ISOLATION VALVES
4. SELECTOR VALVES
5. CONTROL VALVES
6. REGULATORS
7. THRUSTERS
8. CHECK VALVES
9. PRESSURE SWITCHES
10. DISCONNECTS
11. NON-PROPULSIVE NOZZLES
12. GAS GENERATORS
13. TURBOPUMPS
14. HEAT EXCHANGERS
15. ACCUMULATORS
16. FLOW CONTROLLERS

THERMAL (T)

FUEL CELL (TFC)

1. RADIATOR
2. PUMP
3. ISOLATIO

MULTILAYER INSU

1. LOX TANK
2. LH$_2$ TANK
3. FEED LIN
4. INSULATI
5. APS TANK

PURGE BAG (TPB)

1. VENT VAL
2. PURGE VA
3. PURGE LI
4. MANIFOLD
5. SELECTOR
6. PRESSURE

PLUME IMPINGEME

HARDWARE CONDIT

1. COLD PLA
2. HEATERS

Figu

SPACE TUG

AVIONICS (A)

ATA MANAGEMENT (ADM)

1. DIGITAL COMPUTER
2. DATA ACQUISITION UNIT
3. MEASUREMENT PROCESSOR UNIT
4. INTERFACE UNIT
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   3. PUMP
   4. REGULATORS
   5. CONTROL VALVES
   6. CONDENSER
   7. WATER TANK
   8. CHECK VALVES
   9. VENT VALVES
   10. SCRUBBERS
   11. EJECTORS
   12. GAS SEPARATOR
2. BACKUP BATTERY
3. POWER TRANSFER SWITCH
4. POWER BUS MODULE
5. RESISTOR & DIODE MODULE
6. POWER CONTROL SWITCH

RENDEZVOUS AND DOCKING (ARD)

1. LAZER RADAR
2. TELEVISION CAMERA
3. RENDEZVOUS TRANSPONDER
4. OMNI ANTENNA
5. POWER DIVIDER

PROPULSION (P)

MAIN PROPULSION (PMP)

1. LOW SPEED INDUCER
2. TURBOPUMP
3. ISOLATION VALVE
4. PREBURNER
5. CONTROL VALVES
6. REGULATOR
7. RECEIVER
8. CHECK VALVE
9. HEAT EXCHANGER
10. ENGINE

PROPELLANT FEED FILL & DRAIN (PPF)

1. THERMAL ISOLATION SEGMENT
2. PREVALVES
3. FILL & DRAIN VALVES
4. DISCONNECTS
5. J-T EXPANSION VALVES
6. REGULATORS
7. CONTROL VALVES

PROPELLANT MANAGEMENT (PPM)

1. LEVEL SENSORS
2. SENSOR SUPPORT MAST
3. CAP PROBE

PROPELLANT ORIENTATION (PPO)

1. AUXILIARY PROPELLANT TANKS
2. PROPELLANT LEVEL SENSORS

THRUST VECTOR CONTROL (PTC)

1. PUMP MOTOR ACTUATOR
2. RESERVOIR/ACCUMULATOR ACTUATOR

PRESSURIZATION (PPR)

1. CHECK VALVE
2. REGULATORS
3. DIFFUSERS
4. DISCONNECTS
5. CONTROL VALVES

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1. RELIEF VALVES
2. VENT VALVES
3. PRESSURE SWITCHES
4. SELECTOR VALVES
5. CONTROL VALVES
6. DISCONNECTS
7. CHECK VALVES
8. NON-PROPULSIVE NOZZLES

REACTION CONTROL (PRC)

1. BURST DISCS
2. SQUIB VALVES
3. ISOLATION VALVES
4. SELECTOR VALVES
5. CONTROL VALVES
6. REGULATORS
7. THRUSTERS
8. CHECK VALVES
9. PRESSURE SWITCHES
10. DISCONNECTS
11. NON-PROPULSIVE NOZZLES
12. GAS GENERATORS
13. TURBOPUMPS
14. HEAT EXCHANGERS
15. ACCUMULATORS
16. FLOW CONTROLLERS

THERMAL (T)

FUEL CELL (TFC)

1. RADIATORS
2. PUMP
3. ISOLATION VALVE

MULTILAYER INSULATION (TMI)

1. LOX TANK
2. LH_2 TANK
3. FEED LINE
4. INSULATION SUPPORTS
5. APS TANKS

PURGE BAG (TPB)

1. VENT VALVE
2. PURGE VALVE
3. PURGE LINES
4. MANIFOLD
5. SELECTOR VALVE
6. PRESSURE SWITCH

PLUME IMPINGEMENT INSULATION (TPI)

HARDWARE CONDITIONING (THC)

1. COLD PLATES
2. HEATERS

ORBITER (O)

SAFING (OS)

1. RECEIVERS
2. ISOLATION VALVE
3. DISCONNECT
4. RELIEF VALVE

INSULATION REPRESSURIZATION (OIR)

1. RELIEF VALVE
2. RECEIVERS
3. ISOLATION VALVE
4. CHECK VALVE
5. DISCONNECT
6. REGULATOR

DATA MANAGEMENT (ODM)

1. STATUS CONTROL PANEL
2. DISCONNECTS

TUG/SHUTTLE SUPPORT (OTS)

1. FRAMES
2. SIDEWALL
3. LOADS EQUILIZER MECHANISM
4. SYSTEM SUPPORT

DOCKING STRUCTURE (ODS)

1. LATCHING MECHANISM
2. DOCKING PROBE (TUG)

INTERFACE CONNECTIONS (OIC)

1. ELECTRICAL DISCONNECTS
2. FLUID DISCONNECTS
3. UMBILICAL PLATES
4. ACTUATOR MECHANISMS

Figure 1.5-2 Hardware Tree

wiring and component mounting hardware. The tree also lists as a separate category those items which are chargeable to the Tug but remain in the Shuttle orbiter.

The tree outlines the level of hardware detail investigated in the design analysis as well as the breakdown employed.

The design definition studies are presented in two volumes. This volume include the definition of Insulation Subsystems; Meteoroid Protection Subsystem; Structures; Mass Properties; Ground Support Equipment; Reliability; and Safety.

REFERENCES

1. Tug Point Design Study Plan, October 25, 1971 P.D. Office, Program Development, MSFC

2. RST Study, MSC - Pre-Phase A Study for an Analysis of a Reusable Space Tug NAS9-10925; Report SD 71-291-1 thru -6, 22 March 1971

3. NR OOS Study, SAMSO - Orbit-to-Orbit Shuttle (Chemical) Feasibility Study (OOS) Contract F04701-71-C-0171, 4 October 1971, SD 71- -1 thru -7; AF No. SAMSO-TR-71-238, Vol. I-VII

4. MDAC OOS Study SAMSO - Orbit-to-Orbit Shuttle (Chemical) Feasibility Study, Contract F04701-71-C-0173, SAMSO-TR-71-221

5. European Space Tug System (Pre-phase A) - MBB Group: BAC, SNIAS, ELLIOTT, SELENIA, ETCA

6. European Space Tug System (Pre-phase A) HSD Group: SLL, BELL, CAG, OSGMBH, ERNO, FIAT, FVW, SDD, SAEM, MSPA

6.0 INSULATION SUBSYSTEM

6.1 INTRODUCTION

The objective of this study is to design passive insulation systems for the liquid hydrogen and oxygen tanks for repeated use in space environment that will provide needed thermal protection yet exhibit minimum impact on vehicle payload delivery capability.

Much investigation has been conducted by both government and industry in recent years in the field of lightweight high performance insulation systems. (References 6-3, 6-4) As a result, several system choices are available that provide acceptable thermal performance. These systems are capable of reducing boil-off losses by 3 to 4 orders of magnitude when compared to the plastic foams and purged honeycomb insulations currently utilized on vehicles with short operational lives. All systems utilize concentric layers of thin plastic films with metallized reflective surfaces to inhibit the transfer of heat by radiation. The differences between systems are the techniques used to maintain shield separation and the design of attachment posts, joints, and penetrations.

A spacerless configuration utilizing purged MLI was selected as baseline design for this study with concurrance of the NASA. It must be noted that when final design is started advantage will be taken of data to be acquired from new large scale cryogenic tests which are expected to be completed during 1972.

Prior studies at NR have been concentrated on the NARSAM concept with the desire to fill a gap in the technology involving low weight spacerless multi-layer insulation (MLI) systems. This concept utilizes embossed mylar film, aluminized on one side only for reflective shields. The embossment provides separation between adjacent shields without the use of insulative separators. This concept is competitive in thermal performance with other concepts and had a lower installed weight. It is simple to manufacture and install and is easily repaired.

The MLI system selected for this study was patterned after the NARSAM concept but utilizing kapton in place of mylar for the reflective shield material.

6.2 DESIGN REQUIREMENTS AND GUIDELINES

To enable a realistic approach in the insulation system design, study guidelines were expanded to form the requirements charts shown in Figure 6.2-1, 6.2-2 and 6.2-3. Environmental data, were not given by NASA study guidelines, was taken from Reference 6.9-5 and 6.9-6. Ascent and reentry profiles and acceleration loads were taken from prior technology studies, Reference 6.9-3.

CONSIDERATION	GROUND OPERATIONS AMBIENT	GROUND OPERATIONS CRYOGENIC	ASCENT	ORBIT	RE-ENTRY	STORAGE
BOILOFF	- - -	LH_2 & LO_2 10% per hour MAXIMUM	SEE ORBIT	MAX - 3 lb/hr (LO_2 + LH_2) for 168 hour	- - - -	- - - -
SINK TEMPERATURE	28 F Minimum 99 F Maximum	-100 F Minimum +120 F Maximum	-100 F Minimum +200 F Maximum	410 R	-100 F Minimum +200 F Maximum	21 F Minimum 115 F Maximum
PRESSURE	15.0 PSIA MAX. 14.4 PSIA MIN.	See Ambient	Compartment Press. Equal to Profile Ambient	14.7×10^{-13} PSIA	COMPARTMENT Press. Equiv. to Profile Ambient	15.0 PSIA Max. 14.4 PSIA Min.
PURGE SYSTEM	0.1 PSIA Above Compartment	See Ambient	Not Required	Not Required	Not Required	See Ground Ambient
PRECONDITION SYSTEM	1 Max.Temp + 140F 2 $\frac{1}{2}$ SCFM/Ft^2 MLI 3 0.01 PSI Max. across MLI	- - - -	- - - -	- - - -	- - - -	See Ground OPNS
VENT SYSTEM	Interlock with tank pressure to preclude compressive load on tanks.		1 Maintain Δ P less than 0.1 PSI. 2 10^{-3} TORR in 1hr.	- - - -	- - - -	See Ground OPNS
REPRESSURIZATION SYSTEM	NOT REQUIRED	NOT REQUIRED	NOT REQUIRED	NOT REQUIRED	Press. MLI to 0.1 PSI Max. above Comp.	NOT REQUIRED

Figure 6.2-1 Insulation System Design Requirements

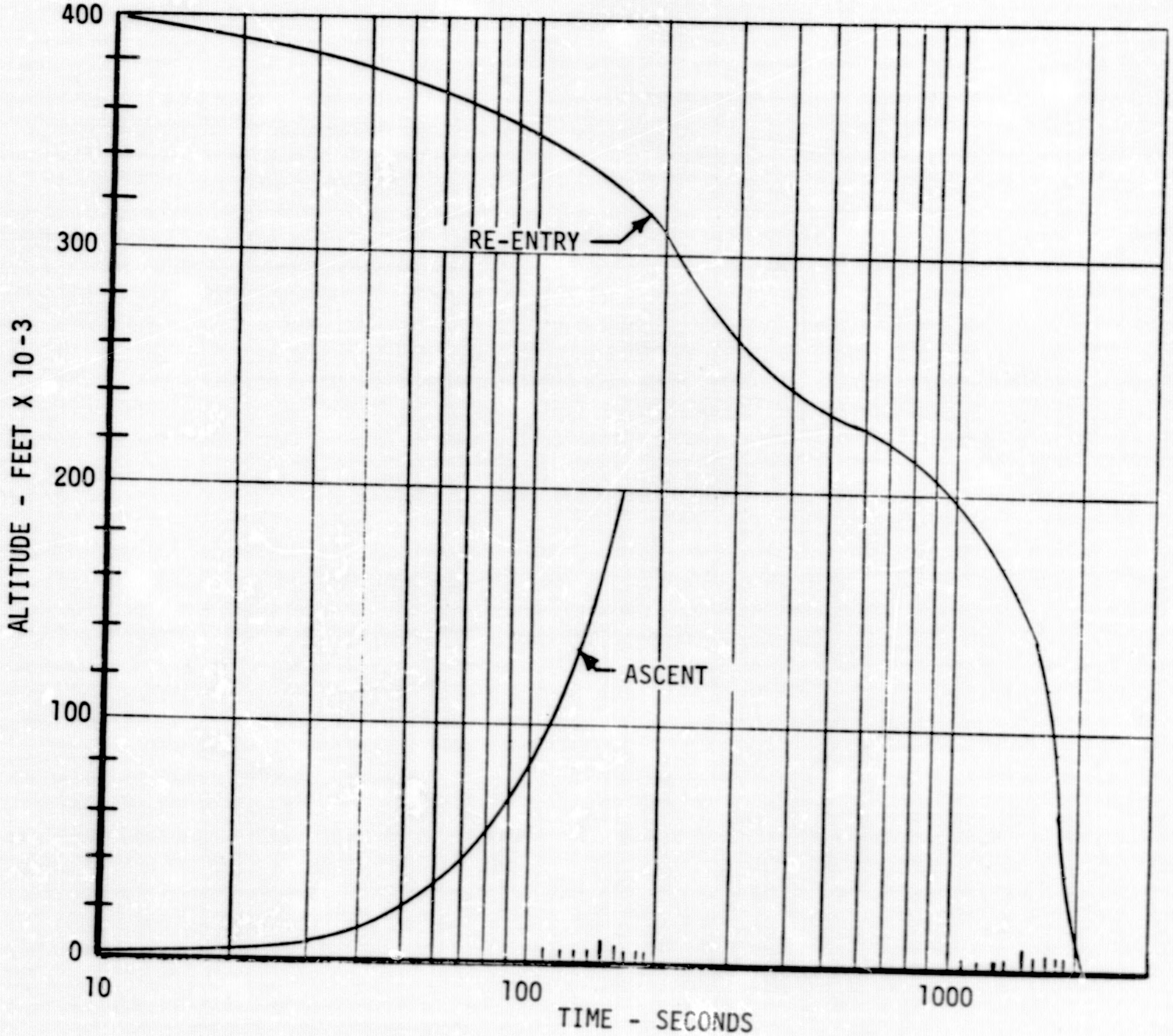
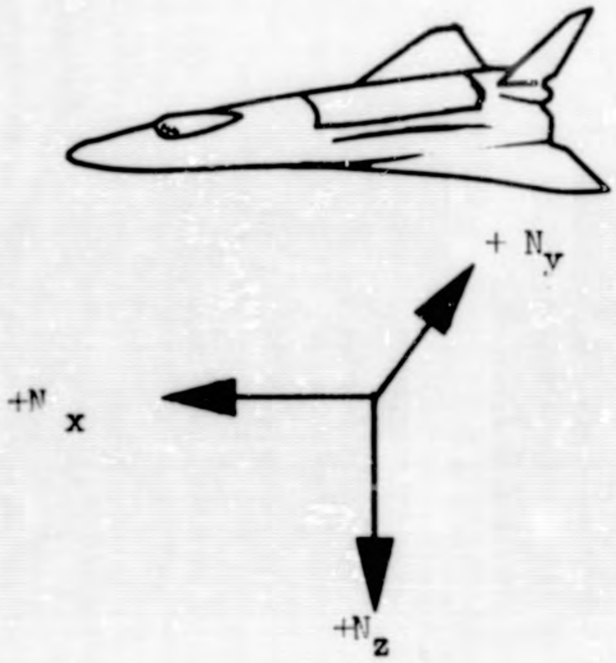


Figure 6.2-2 FLIGHT PROFILES



Condition	N_x (g)	N_y (g)	N_z (g)
Launch	1.4 $\pm$1.6	$\pm$ 1.0	$\pm$ 1.0
High Q Booster Thrust	1.9 $\pm$0.3	$\pm$ 1.0	0.8 $\pm$ 0.2
End Boost (Booster Thrust)	3 $\pm$ 0.3	$\pm$ 0.6	$\pm$ 0.6
End Burn (Orbiter Thrust)	3 $\pm$ 0.3	$\pm$ 0.5	$\pm$ 0.5
Orbiter Entry	- 0.5	$\pm$ 1.0	- 3.0 $\pm$ 1.0
Orbiter Flyback	- 0.5	$\pm$ 1.0	+ 1.0 -2.5 $\pm$1.0
Landing	- 1.3	$\pm$ 0.5	- 2.7 $\pm$ 0.5

Figure 6.2-3 Structural Loads

6.3 BASELINE INSULATION CONCEPT DESCRIPTION

Based on analysis presented in Section 3.2.1, of Vol. II, thirty shields are required to provide necessary thermal protection. The design employs the natural lay concept in the configuration shown in FIGURE 6.3-1. The MLI and ancillary subsystem concepts are described in the following paragraphs. Design details are discussed in Section 6.5.

6.3.1 Multi-Layer Insulation System (MLI)

Separation of the reflective shields and installation of the shields on the tanks in a manner that minimizes direct conduction is a major concern in selection of the MLI system. For the selected system, the lowest thermal conductivity is realized when layer density is between approximately 40 and 50 shields per inch of insulation thickness. If the density is increased, thermal performance is proportionately degraded; therefore, the design must incorporate provisions to ensure that the layer density does not increase as a result of external effects during cryogenic operations.

The natural lay concept was selected for this design to preclude such occurrence. This concept takes advantage of the natural separation afforded by the embossment. The shields, separated by the random peaks in the embossment pattern, assume a natural density when stacked as shown in Figure 6.3-2. Compressive loads or tensile loads, as shown in Figures 6.3-2 which increase layer density, can only be caused by external forces such as weight, acceleration, pressure, or thermal contraction and must be reacted at the attach posts. In the natural lay concept, external loads are eliminated by providing slack in the MLI between posts, as shown in Figure 6.3-2. This offsets the difference in thermal contraction between the MLI and aluminum tank and precludes transfer of loads due to weight beyond the post that directly supports it. To prevent density increase due to MLI "bunching," the individual gores are cut in a manner that allows them to drape smoothly over the bicontour surface without wrinkling. Gore width to accomplish this is empirically determined as shown in Figure 6.3-3.

Because of the fragile nature of the shields, five individual shields are stacked together to form modules which are joined together on small cut-off tabs. This enables manufacturing personnel to handle them conveniently and without damage. In this design, the insulation is made up of six module layers with five shields in each module. Each layer of modules overlap the joints of the layer below it to minimize heat loss.

The shield material is perforated with small holes totaling approximately 2 percent of the total area. This permits gasses to flow radially through the MLI. Tests at NR have shown that 2 percent perforations will limit pressure differential to less than 0.0108 psi with approximately one SCFM of purge gas per square foot flowing through 90 shields. For the 30 shields required in this study, the pressure difference will be considerably less. In final design, this will be evaluated to determine advantages of reducing perforation area and accepting the slight increase in pressure as trade for reduced heat loss.

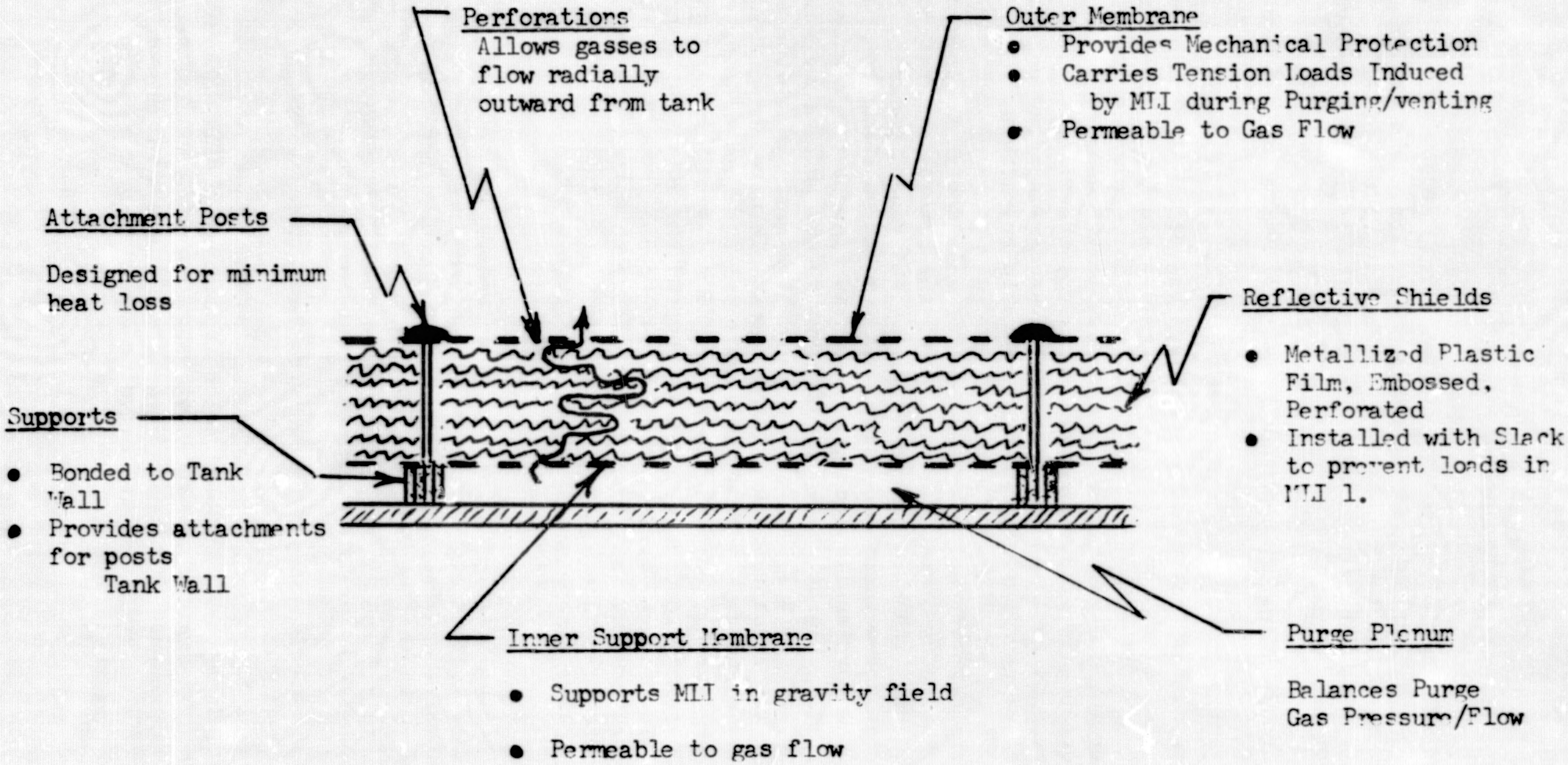
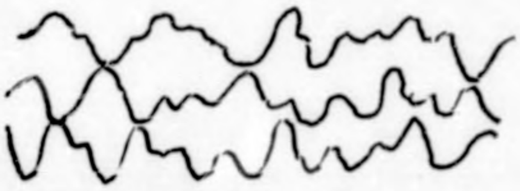
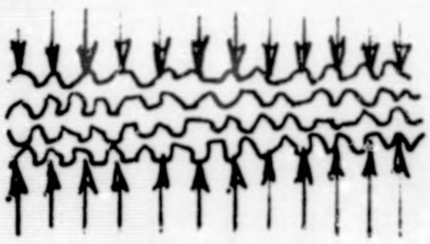


Figure 6.3-1 Natural Layup Concept

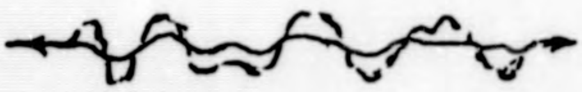


(a) Natural layup density

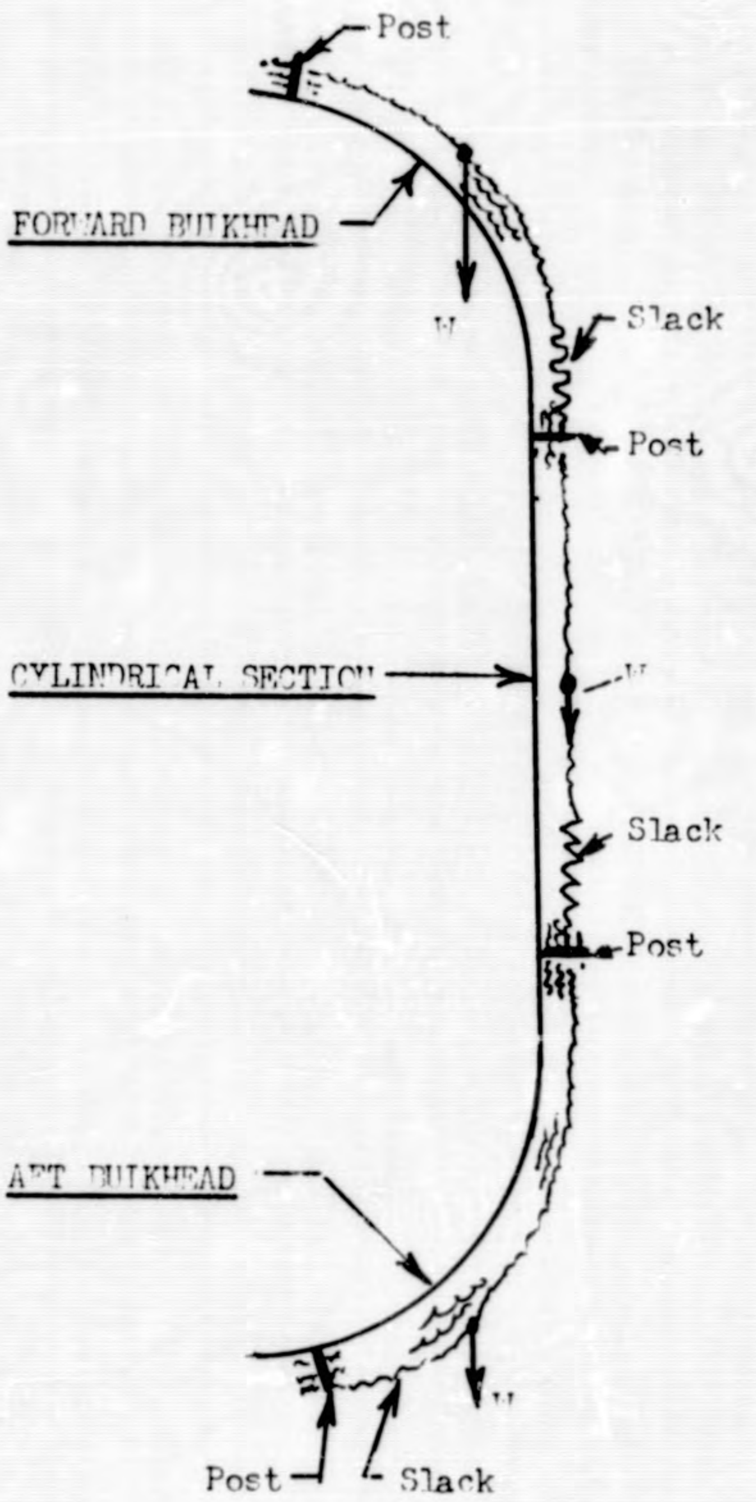
- Established by depth of embossment.
- Adjacent shields make contact at random points.



(b) Compressive loads increase contact pressure between adjacent shields - caused by
- Weight
- Acceleration
- Dynamic environment
- Pressure



(c) Tensile loads decrease embossment depth - caused by
- Thermal contraction
- Pressure



(d) LOAD DISTRIBUTION TO SUPPORT POSTS

Figure 6.3-2 Insulation Load Concepts

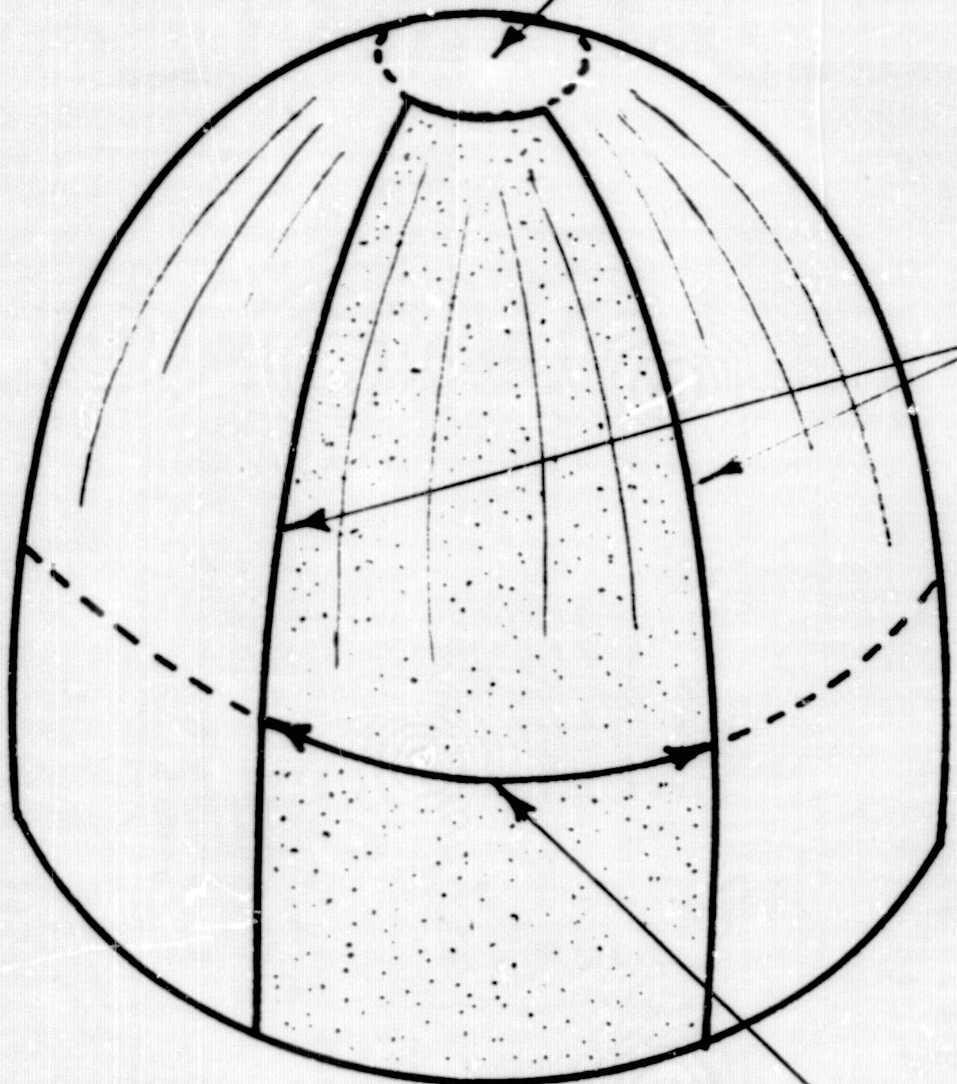


Figure 6.3-3 Maximum Width of Reflective Shield

6.3.2 Ancillary Systems

Ancillary systems are required for the MLI, both ground support and airborne. These systems are defined in FIGURE 6.3-4 and described in the following paragraphs.

Purge System

The purpose of the purge system is primarily to isolate the MLI from the atmospheric environment to preclude introduction of contaminates such as moisture, chemicals or dust that might cause corrosion or otherwise degrade thermal performance. The purge gas may be dry GN_2 or air and is introduced into the Tug structure at pressures not in excess of 0.1 psig. It must be noted that purging is not required while the orbiter compartment conditioning system is in operation.

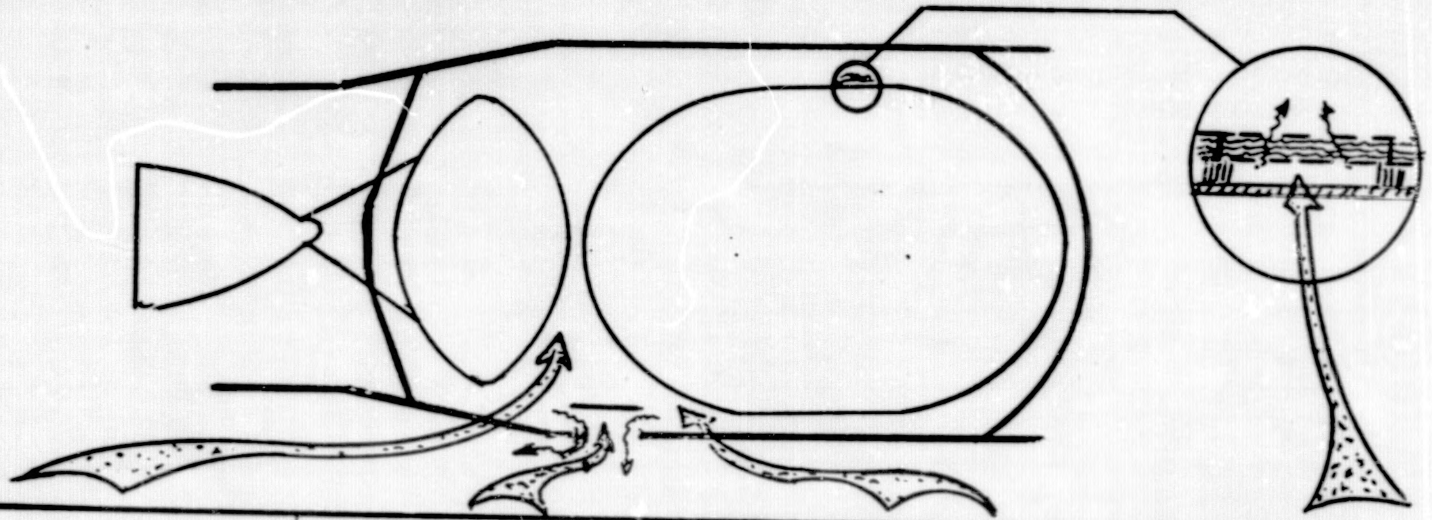
Prior to loading cryogens, it is necessary that the GN_2 or air be flushed from the MLI with helium. This prevents condensation and freezing of the condensable gasses which might otherwise outgas in the space vacuum with resultant degradation in thermal performance. The helium is introduced into the inner plenum through the purge manifold. It flows radially outward through the MLI, replacing the GN_2 or air which is subsequently vented into the orbiter compartment.

Prior tests at NR demonstrated that helium pressure of 0.10 psid will effectively inhibit both intrusion and back diffusion of ambient atmosphere through holes of identifiable size. Helium purge is maintained within Tug structure at all times while cryogens are loaded and is disconnected by umbilical pull-back prior to launch.

Vent System

During ascent, the orbiter compartment pressure decreases with flight altitude. Two vent valves, located in the Tug structure are opened at launch to permit the residual helium purge gas within the Tug structure and MLI to evacuate into the orbiter compartment. The valve apertures are sized to permit evacuation at a rate that will preclude pressure differential across the Tug structure from exceeding 0.1 psi. A single valve having a 5-inch diameter opening is satisfactory; however, two valves have been incorporated for redundancy. Recent tests on NARSAM materials show that when they are conditioned and protected in accordance with existing NR specifications, the outgassing rates in the molecular flow regime are considerably less than previously believed and, therefore, the valve apertures are sized by flow in the laminar regime as was done in this study.

The vent valves remain open during orbital operations to implement outgassing of moisture and other volatile products. In addition, the valves are interlocked by pressure switches at all times to ensure that the Tug structure is not over pressurized and that the cryogen tanks are not subjected to collapsing pressures.



	PURGE	VENT	REPRESSURIZATION	PRE-CONDITIONING
SYSTEM	MAINTAIN PRESSURE INSIDE TUG STRUCTURE ABOVE AMBIENT PRESSURE (0.1 PSI)	EVACUATE RESIDUAL HELIUM PURGE AND MATERIAL OUTGAS PRODUCTS (MAX Δ P =0.1 PSI, 10^{-3} TORR IN ONE HOUR)	MAINTAIN PRESSURE INSIDE TUG STRUCTURE ABOVE AMBIENT PRESSURE	FLOW DRY PURGE GAS BROADSIDE THROUGH MLI (MAX PRESS = 0.01 PSI 0, MAX TEMPERATURE + 140 F)
PURPOSE	GROUND-AMBIENT (N_2/DRY AIR) PREVENT INTRUSION OF MOISTURE AND ATMOSPHERIC CONTAMINATES GROUND-CRYOGENIC (GH_e) PREVENT INTRUSION OF CONTAMINATES AND CONDENSABLE GASSES.	LAUNCH • PREVENT OVERPRESSURE OF TUG STRUCTURE DURING ASCENT. • ACHIEVE LOW MLI PRESSURE RAPIDLY TO ENHANCE THERMAL PROTECTION. ORBIT MAINTAIN LOW MLI PRESSURE FOR OPTIMUM THERMAL PERFORMANCE.	REENTRY (H_E) • PREVENT INTRUSION OF ATMOSPHERIC CONTAMINATES. • PROVIDE HEAT SHORT BETWEEN OUTER MLI SHIELDS AND TANK WALL TO PRECLUDE OVERHEAT OF SHIELDS.	GROUND AMBIENT (N_2 OR DRY AIR) REMOVE MOISTURE ACCIDENTALLY INTRODUCED INTO MLI DURING STORAGE, REFURBISHMENT OR PRE-LAUNCH OPERATIONS.

Figure 6.3-4 Purge and Vent Systems

Repressurization System

During re-entry the orbiter compartment is repressurized with ambient atmospheric air. This might introduce (1) collapsing pressures on the Tug structure and (2) contaminates which may subsequently degrade reflective surfaces in the MLI. To overcome these problems, the Tug structure is pressurized with helium at a rate that maintains a differential pressure of 0.1 psi above compartment pressure.

It is important to note that the LH_2 and LO_2 tanks are vented prior to re-entry and that the tank walls can be expected to be warmer than 300R due to the inerting purge. Tug structure temperatures will also increase to +200 F during re-entry which would subsequently be experienced by the MLI. By introducing the helium at pressure greater than 10^{-2} TORR at start of reentry a thermal short is provided between the outer reflective shields and the tank wall, thereby limiting the temperature of the outer shields. No attempt was made in this study to optimize the repressurization schedule. This will be done in the final program design to ensure that the system weight is minimum and that maximum protection is afforded if required by the shield material.

Pre-Conditioning System

The basic design procurement plan requires that the insulation system material by vacuum dried by the supplier as the final processing operation and that it be shipped in humidity controlled containers. During manufacture it is processed in environmentally controlled areas and after installation it is protected by dry pressurized environment at all times.

The purpose of these controls is to preclude intrusion of moisture by climatic changes. However, based on operational experiences of prior programs, such as Saturn II, it is naive to assume moisture will not be inadvertently introduced during field operations due to severe weather, accident, or support equipment failures.

A preconditioning system has been included to remove such moisture from the MLI. Dry purge gas, either N_2 or air, is introduced through the manifold into the plenum below the MLI inner support membrane as shown in Figure 6.3-4. It flows radially outward in a labyrinthian path through the perforations and the interstitial areas between the shields. The moisture is mechanically agitated, absorbed and removed by the dry purge.

It is anticipated that a series of sensors will be installed in key locations in the MLI to identify the presence of moisture and to indicate when it has been successfully removed. The availability of such sensors is anticipated within the 1976 state of the art.

6.4 DESIGN CONFIGURATION SELECTION

6.4.1 Insulated Areas

Insulation concepts for each of these areas have been defined and are presented in the referenced engineering drawings. The Top Assembly Drawing, V7-923610 (Figure 6.4-1) shows the location of each insulation system. The following areas of the Space Tug require multi-layer insulation (MLI):

a. LH_2 tank (Ref. Dwg. V7-923612), Figure 6.4-3

b. LOX tank (Ref. Dwg. V7-923613), Figure 6.4-4

c. LH_2 feedline and fill/vent line (Ref. Dwg. V7-923614), Figure 6.4-5

d. LOX feedline and fill/vent line (Ref. Dwg. V7-923614), Figure 6.4-5

e. Reaction control system (RCS) accumulator tanks (Ref. Dwg. V7-923617), Figure 6.4-6

6.4.2 Weight Constraints

The maximum allowable insulation system weight of 253 lbs was targeted by study guidelines. Included in this total are the multi-layer insulation, internal and external MLI support provisions, support posts, seals, and purge system supply lines and manifolds. The weight limit does not include helium bottles and their supports, and plumbing that will not be carried in the Tug but will remain in the Shuttle orbiter cargo bay. See Section 9 for detailed weight breakdown.

6.4.3 MLI Design Material Selection

Candidate Materials (Reference Section 6.6)

The following materials were considered for use as reflective shields.

a. Aluminized mylar

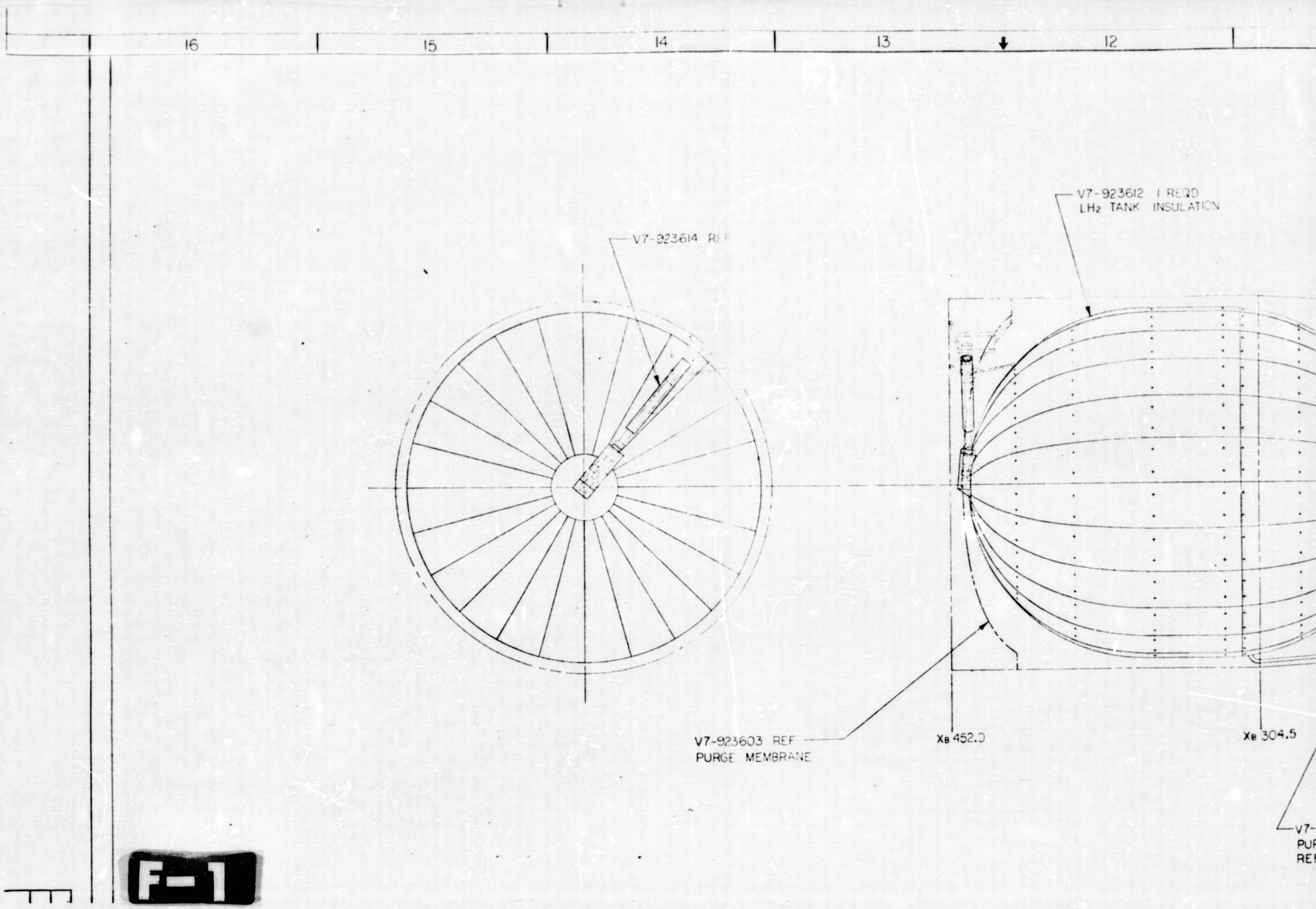
b. Aluminized kapton

c. Goldized mylar

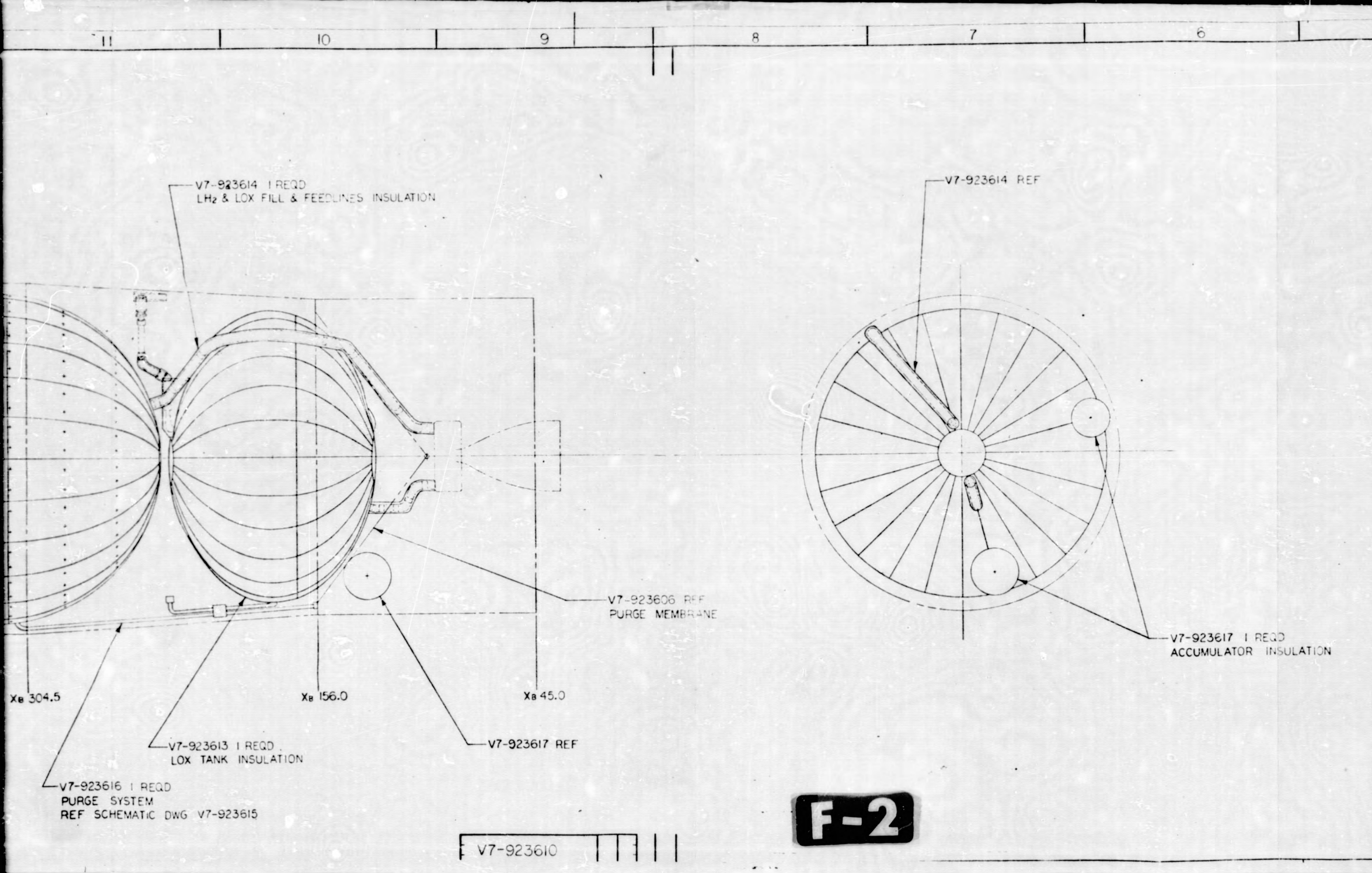
d. Goldized kapton

Each of the candidates must be embossed, perforated, and metallized on one side of the film substrate only. Embossed Mylar (polyester) has been used extensively as an MLI material, whereas Kapton (polyimide) requires further development to fully demonstrate its embossment capability, but is expected to be available as 1976 state of art.

Aluminized Kapton was selected as the MLI material. The rationale for this selection is presented below.



16
15
14
13
12
V7-923614 REF
V7-923612 1 REQD
LH2 TANK INSULATION
V7-923603 REF
PURGE MEMBRANE
XB 452.0
XB 304.5
V7-9
PURG
REF
F-1



11
10
9
8
7
6
V7-923614 1 REQD
LH2 & LOX FILL & FEEDLINES INSULATION
V7-923614 REF
V7-923606 REF
PURGE MEMBRANE
V7-923617 1 REQD
ACCUMULATOR INSULATION
XB 304.5
XB 156.0
XB 45.0
V7-923617 REF
V7-923613 1 REQD
LOX TANK INSULATION
V7-923616 1 REQD
PURGE SYSTEM
REF SCHEMATIC DWG V7-923615
V7-923610

F-2

Figure 6.4-1
Insulation - Multi Layer, Space Tug, Top Assy of

REVISIONS	
ZONE	LTR	DESCRIPTION	DATE	APPROVED

INSULATION — MULTI LAYER, SPACE TUG, TOP ASSY OF

SIZE	CODE IDENT NO	DRAWING NO.
L	03953	V7-923610

SCALE 1/20 H. J. LILLY

V7-923610

F-3

Film Substrate Material

The prime consideration in the selection of the MLI substrate material was the environmental temperature of 200 F experienced during re-entry. This temperature exceeds the maximum allowable of 140 F for embossed Mylar under compressive loading but is well within the capability of Kapton.

It is pointed out that while it is possible to limit the maximum temperature of the MLI during re-entry by repressurization with helium which would have permitted the use of embossed Mylar, Kapton, also permits a wider range of alternatives with respect to subsystem selections, propellant tank venting, and orbiter interfaces.

Metallized Coating

The following factors were considered in the selection of the metallized coating (gold vs. aluminum):

a. Thermal performance

b. Corrosion resistance

c. Producibility

d. Cost

Aluminum was selected on the basis of state-of-the-art producibility and reduced cost. Aluminum and gold both meet the thermal performance requirements of the Tug mission. Since aluminized material may be susceptible to moisture corrosion, a requirement for continuous environmental control for the MLI is recognized. However, some control is required regardless of the metal selected, to preclude intrusion of excessive moisture with subsequent adverse effect on the MLI vent-down rate. Control is also required for protection of the cryogenic tanks.

The RCS accumulator tanks and transfer lines, are located in an uncontrolled area (Ref. Dwg. V7-923617) without protection of the purge and preconditioning systems. The aluminized Kapton was selected here, too, for uniformity plus reduced thermal penalty in early flight. It can be easily re-insulated in the event of accidental moisture intrusion.

Alternate MLI Material

Since aluminized Mylar has been tested and proven adequate, it would be desirable to use this material, in lieu of developing aluminized Kapton. Mylar could be used if the maximum environmental temperature of the MLI were reduced. As discussed earlier this could be accomplished by the introduction of helium to repressurize the MLI system prior to re-entry, thus creating a heat short to the propellant tank. If the tank is sufficiently cold at the time of repressurization, the MLI temperature may be lowered to an acceptable level during re-entry. When final design is undertaken, analysis will be made in these areas and consideration given to use of Mylar.

6.4.4 MLI System Concept Selection

Candidate Configurations and Selection Rationale

FIGURE 6.4-2 shows the four configurations that were considered for the MLI system. Configuration 1 utilizes a sealed structure for environmental control with MLI supported from the structure. This concept was eliminated due to the possibility of freezing of the LOX by the LH_2 and possible requirement for heating other subsystem components.

Configuration 2 shows MLI and a sealed purge bag supported from the propellant tanks and transfer lines. This system is considered to be overly complex and has a weight penalty due to the requirement for sealing large surface areas.

Configuration 3 features a sealed structure including a membrane separating the LOX and LH_2 tanks. The MLI is supported from the tanks. Excessive weight due to the separator membrane disqualified the concept.

Configuration 4 is identical to Configuration 3 but without the separator membrane. With this concept, it is possible for excessive leakage from GO_2 and GH_2 flanges to mix. Safety analysis was conducted which showed that the helium purge flow rate is conservative and well able to rapidly dilute combustible mixtures. Excessive leakage can be avoided through proper design of the propellant transfer system. This configuration provides insulation of individual cryogenic components, which isolates LOX and LH_2 temperatures. The sealed structure provides air isolation for the entire MLI system without the weight and installation problems associated with individual purge bags. Configuration 4 was selected as the best trade between system performance, weight, and producibility.

6.4.5 MLI Purge and Conditioning

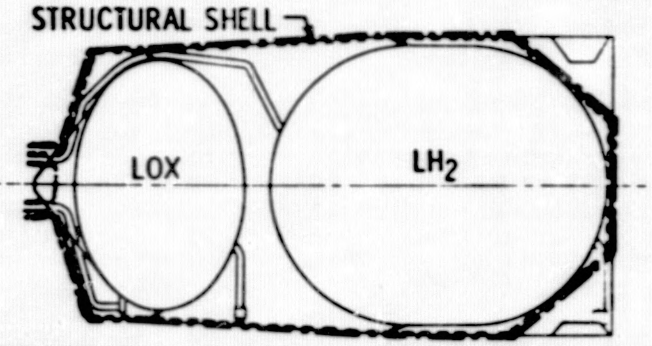
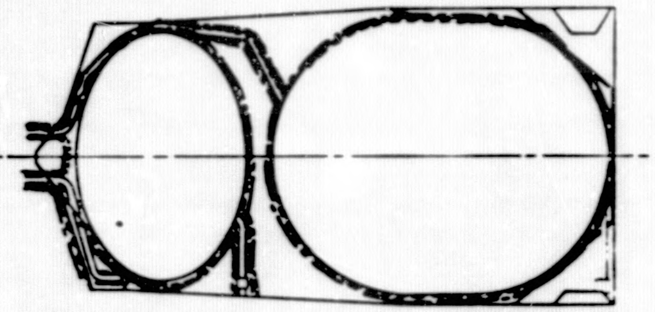
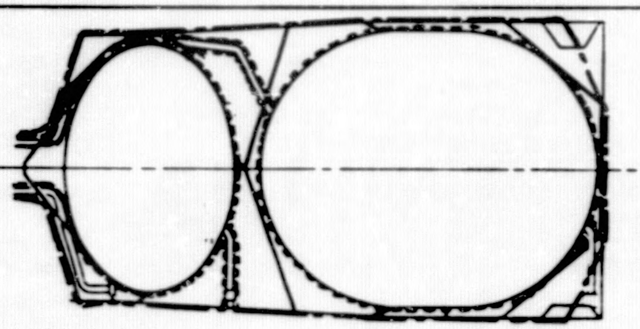
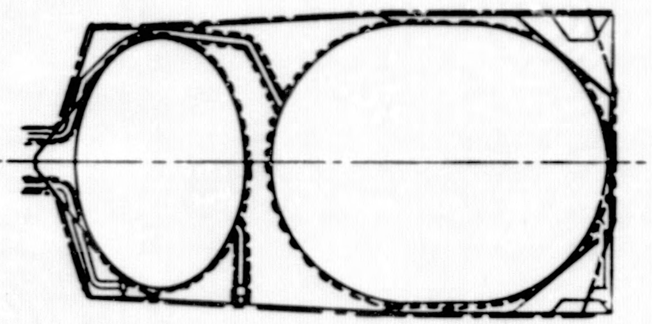
The Space Tug contains two systems for the introduction of purge and conditioning gases to both sides of the MLI (Ref. Schematic Dwg. V7-923615, Figure 6.4-7). An internal manifold mounted on each tank provides purge gas to the backside of the MLI, and a diffuser is used to pressurize the interior of the sealed structure. An internal MLI purge is required for the following reasons:

a. Positive removal of condensible gases

b. Positive removal of entrapped water vapor

c. Effective drying following accidental intrusion of liquid water

d. Balanced repressurization of MLI at re-entry.

CONFIGURATION	DESCRIPTION	ADVANTAGES	DISADVANTAGES
① 	SEALED & INSULATED STRUCTURE	• MINIMUM INSULATION AREA • MINIMUM SEALED SURFACE PURG & REPRESSURIZATION SYSTEM	• LOX SUBJECT TO FREEZING BY LH_2 • RESTRICTED ACCESS TO ONBOARD SYSTEMS
② 	SEALED & INSULATED COMPONENTS	• NO STRUCTURAL SEAL REQUIRED • REDUCED GAS VOLUME FOR PURGE & REPRESSURIZATION • MAXIMUM ACCESSIBILITY OF ONBOARD SYSTEMS	• ELABORATE PURGE & REPRESSURIZATION SYSTEM • SEALS REQUIRE ADDITIONAL SUPPORT POSTS
③ 	ISOLATED SEALED STRUCTURE & INSULATED COMPONENTS	• IMPROVED ACCESS TO INSULATION • CONTAMINANTS ISOLATED TO LIMITED VOLUMES	• SPECIAL SUPPORTS FOR INTER-TANK SEAL • INCREASED SEALED SURFACE AREA • RELATIVELY COMPLEX PURGE & VENT SYSTEM
④ 	COMBINED SEALED STRUCTURE & INSULATED COMPONENTS	• SIMPLIFIED PURGE & REPRESSURIZATION SYSTEM • MINIMUM SEALED SURFACE AREA • IMPROVED ACCESS TO INSULATION	• PROXIMITY OF LH_2 & LOX LEAKAGE SOURCES

CODE: SEAL —·—
MLI -----

Figure 6.4-2 Candidate Configuration

The external purge system is required in order to provide the following:

a. Continuous environmental control for the MLI

b. Capability of removing condensible gases from Tug interior

c. Rapid dilution of a hazardous atmosphere due to excessive GH_2/GO_2 leakage

The main propellant tanks MLI is purged and conditioned in various stages. During storage in an uncontrolled environment, the sealed structure is pressurized with dry air or nitrogen to preclude MLI contamination. Preconditioning of the MLI is accomplished by first flowing warm nitrogen through the internal purge system to remove moisture from the MLI. After the drying operation is completed, the interior of the Tug is continuously pressurized with nitrogen to preclude any further moisture or contaminate intrusion prior to launch.

During pre-launch operations the MLI is purged with ambient helium through the internal purge system. The MLI is thus flushed of condensible gases, notably nitrogen. At this time the helium purge is switched to the compartment diffuser to isolate the MLI from the flowing gas. This purge continues through cryogenic operations until launch. During ascent and while in orbit, all purge systems are inoperative.

Internal purge provisions are made for the propellant vent line insulation, since the vent lines will be functioning prior to launch and during ascent.

The MLI on the RCS accumulator tanks and lines does not require internal purging, since it is subject to the dry nitrogen atmosphere of the orbiter cargo bay; therefore, moisture intrusion is unlikely. The MLI, being easily accessible, would be replaced in the event of a severe moisture condition. Furthermore, the operating temperatures of the accumulator tanks and lines are not low enough to cause nitrogen condensation and any moisture which accidentally intrudes will be evacuated in early flight without penalty to thermal performance.

An internal purge of the propellant feedlines MLI is not required for the following reasons:

a. Volume of residual condensible gases is negligible

b. Volume of possible liquid water intrusion is small and can be removed by external diffusion purge

c. Feedline MLI need only function in orbit; moisture can outgas during ascent

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6.5 MLI SYSTEM DESIGN DESCRIPTION

6.5.1 Propellant Tank Insulation (Ref. Dwg. V7-923612, Figure 6.4-3, V7-923613, Figure 6.4-4.)

The MLI for both the LH_2 and LOX tanks consists of aluminized Kapton film which is embossed and perforated to 2 percent of its surface area. The insulation thickness of 1/2 inch, which amounts to 30 shields of MLI having a layer density of 60 shields per inch. During installation the MLI is handled in modules of five shields which are heat sealed together at edge tabs. The MLI shields are cut in gore segments, each one accounting for 1/24 of the tank circumference. The vertical joints between adjacent gore segment modules are heat sealed or bonded together at installation. These joints are staggered throughout the insulation thickness by successive overlapping of the modules.

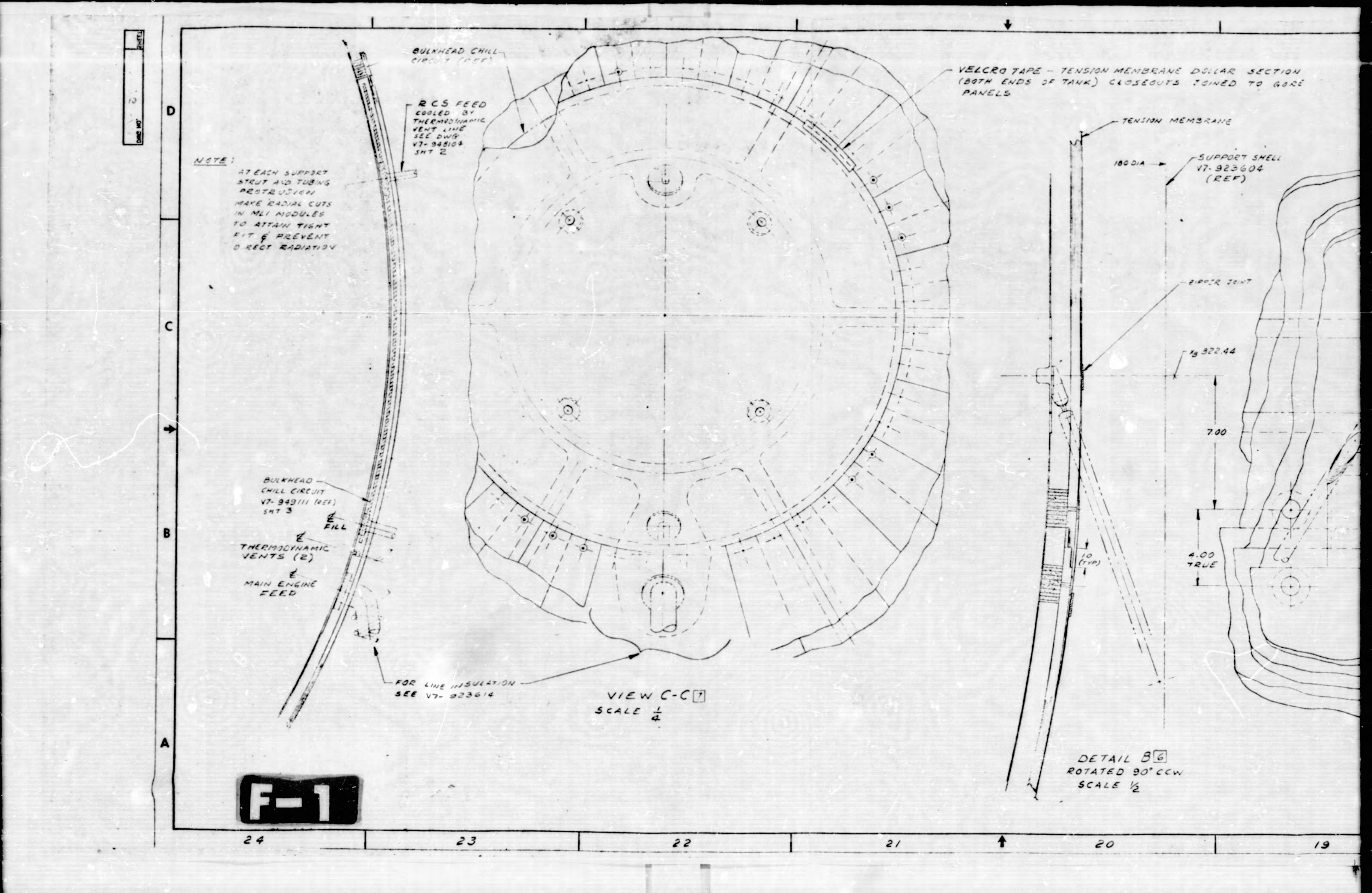
The MLI on each tank contains a single circumferential step joint located just below the tank support struts. This joint facilitates the installation of the MLI around the struts without the use of special plug sections. Since the support struts penetrate the MLI at oblique angles, an elliptical cutout must be made in each shield. Slits running vertically from the cutouts to the circumferential joint or horizontally to the edge of the module permit the shields to be folded back, installed around the struts, and then heat sealed together at the tabs.

Special MLI plugs are provided for the dollar section closeouts on each tank, as well as the tank access doors. Plugs are also used to insulate around the thrust cone support struts on the LOX tank.

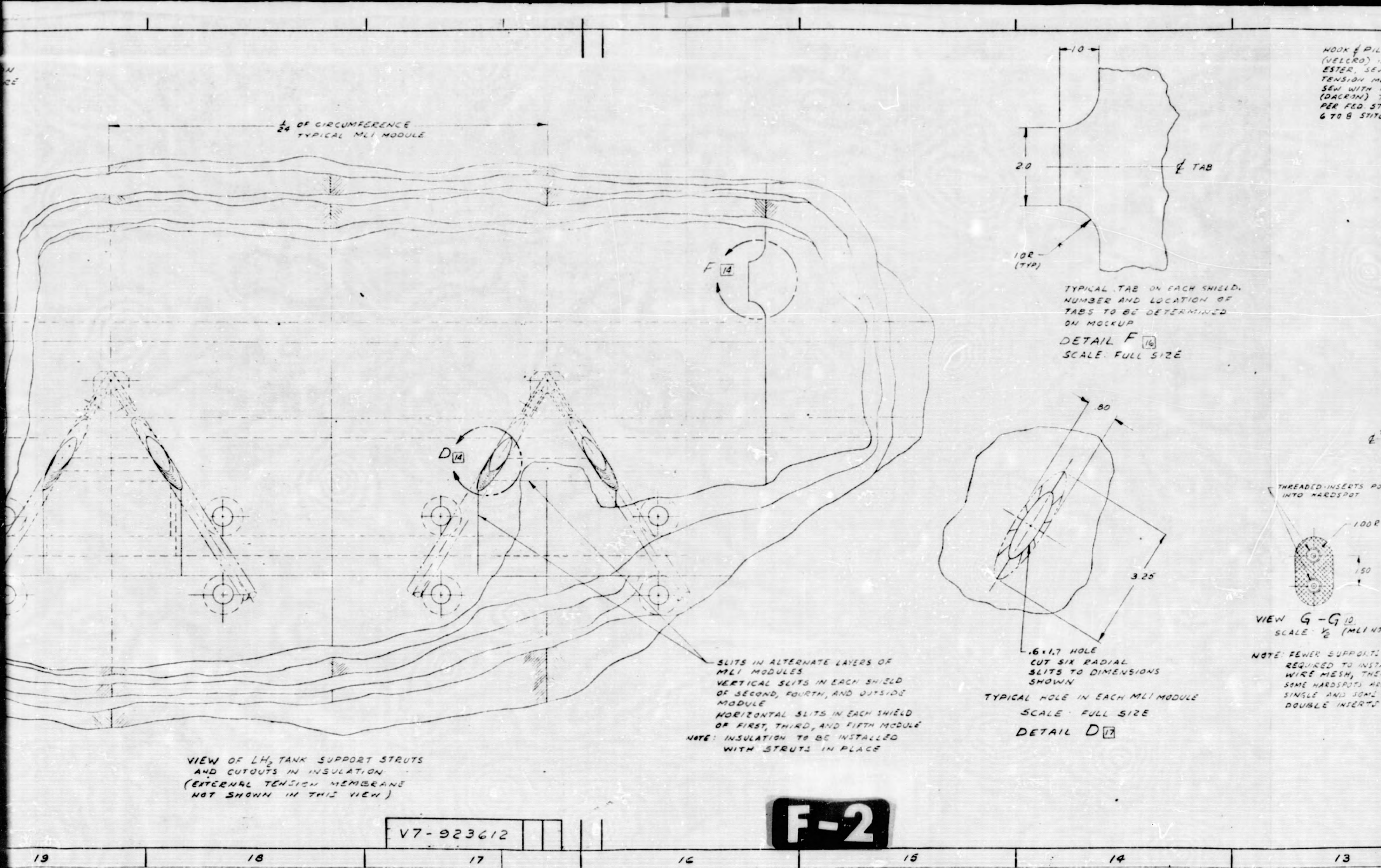
The MLI is supported on the propellant tanks with posts molded from epoxy/fiberglass. The posts are hollow with a wall thickness of 0.010 inches to minimize heat leak. At each post location a foam-filled phenolic honeycomb hardspot is bonded to the tank surface. A threaded insert is bonded into each hardspot, which accepts the MLI support post.

In order to effectively purge the MLI from the back side, a 1.00 inch annulus is provided between the propellant tank and the insulation. The internal purge manifold is located in this annulus. It distributes the gas at balanced pressure through the annulus, permitting even flow through the MLI. The MLI is spaced away from the tank surface on an aluminum wire mesh supported by the foam/honeycomb hardspots. Threaded fasteners are used to attach the mesh to inserts in the hardspots. The mesh is installed in stretch-formed gore segments with cutouts provided for all MLI penetrations. The annulus is sized to assure even distribution of the purge gas and to afford clearance between the MLI and irregular shapes on the tank surface.

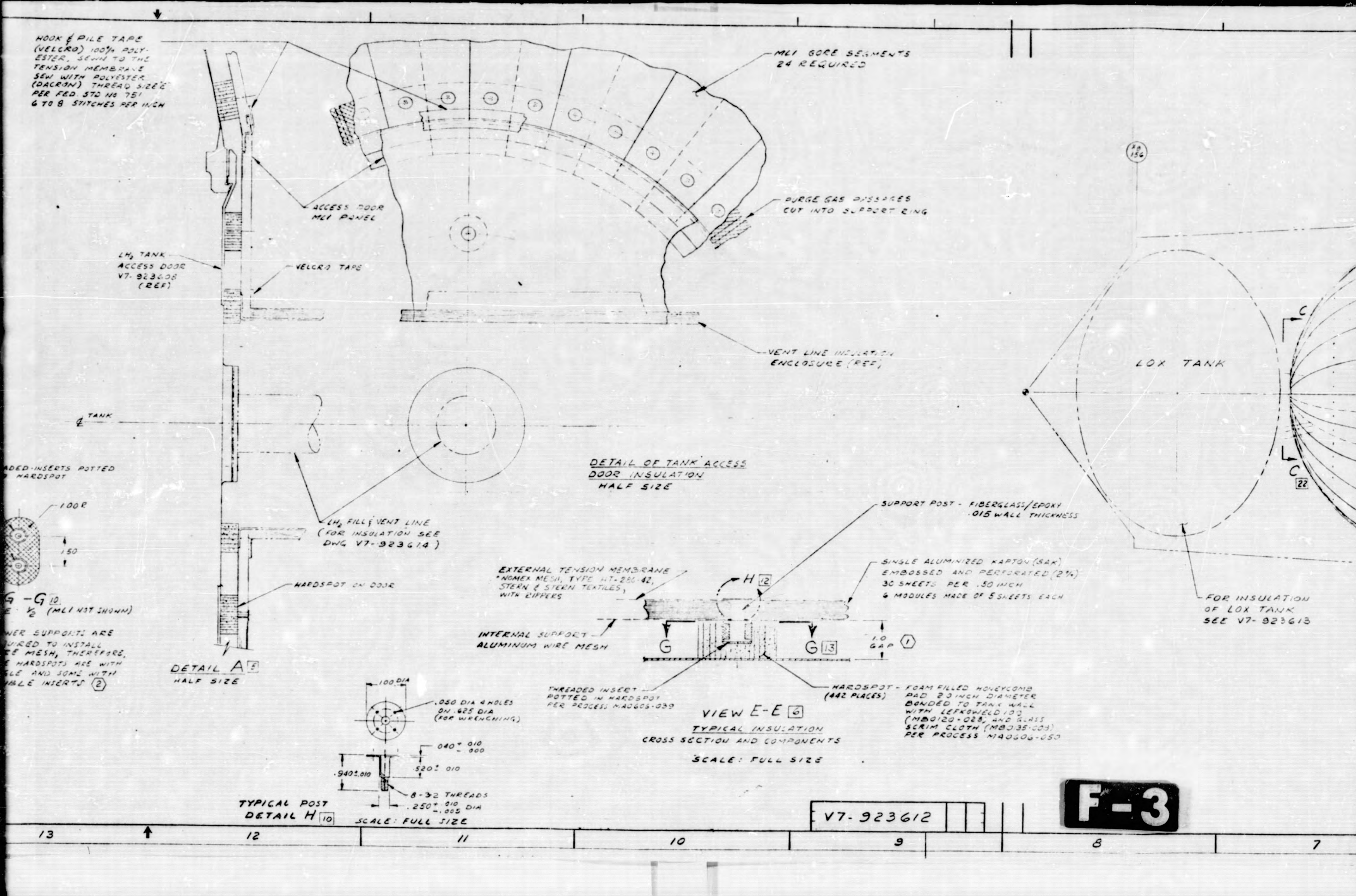
An external tension membrane is provided to contain the MLI during purge and preconditioning operation when backside pressure forces are being exerted, and to continuously protect the insulation surface from mechanical damage.

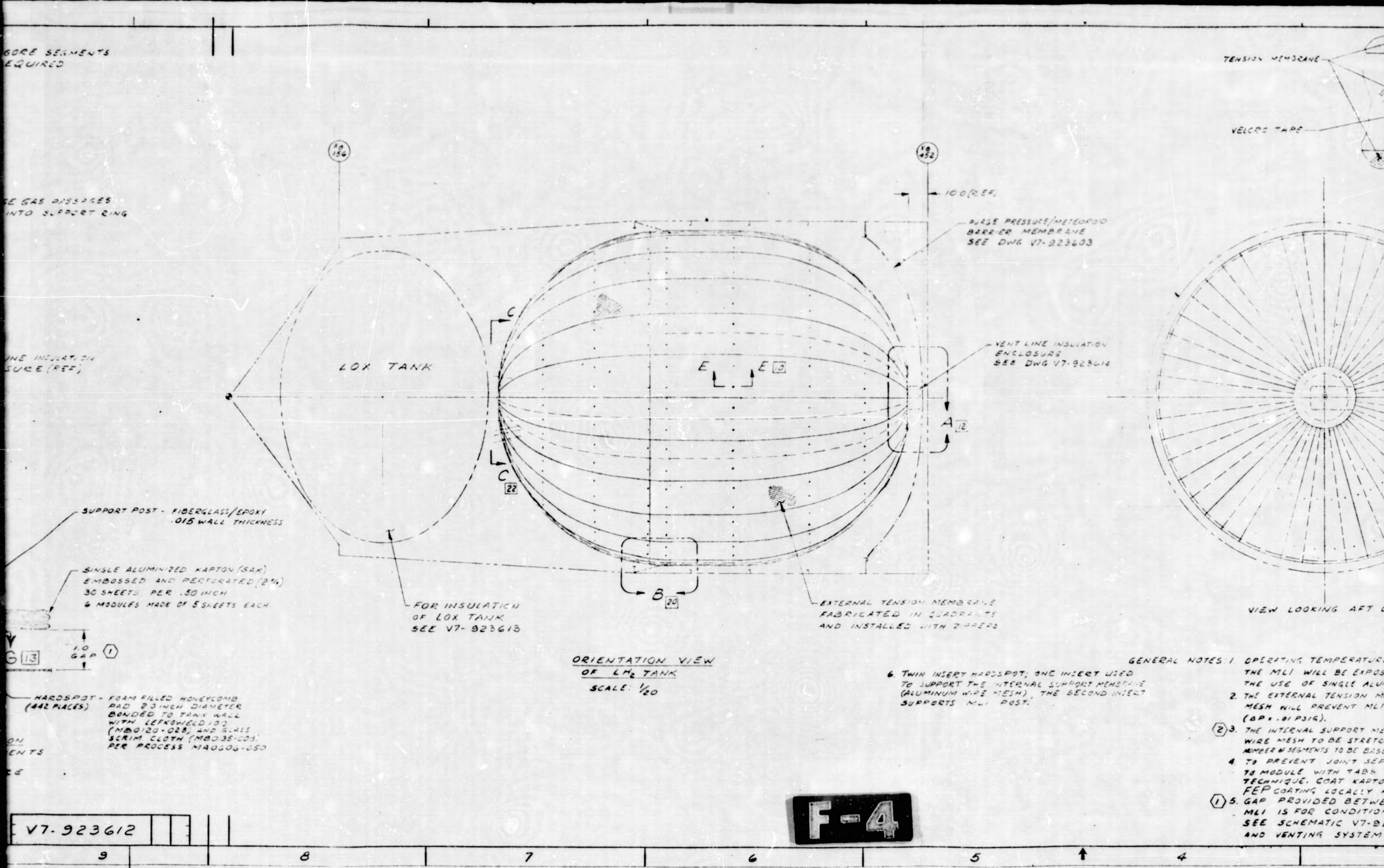


BULKHEAD CHILL CIRCUIT (REF)
VELCRO TAPE - TENSION MEMBRANE DOLLAR SECTION (BOTH ENDS OF TANK) CLOSEOUTS JOINED TO GORE PANELS
RCS FEED COOLED BY THERMODYNAMIC VENT LINE SEE DWG V7-948107 SHT 2
TENSION MEMBRANE
180 DIA
SUPPORT SHELL V7-923604 (REF)
NOTE: AT EACH SUPPORT STRUT AND TUBING PROTRUSION MAKE RADIAL CUTS IN MLI MODULES TO ATTAIN TIGHT FIT & PREVENT DIRECT RADIATION
ZIPPER JOINT
322.44
7.00
4.00 TRUE
.10 (TYP)
BULKHEAD CHILL CIRCUIT V7-949111 (REF) SHT 3
℄ FILL
℄ THERMODYNAMIC VENTS (2)
℄ MAIN ENGINE FEED
FOR LINE INSULATION SEE V7-923614
VIEW C-C 7
SCALE 1/4
DETAIL B 6
ROTATED 90° CCW
SCALE 1/2
F-1
D
C
B
A
24
23
22
21
20
19

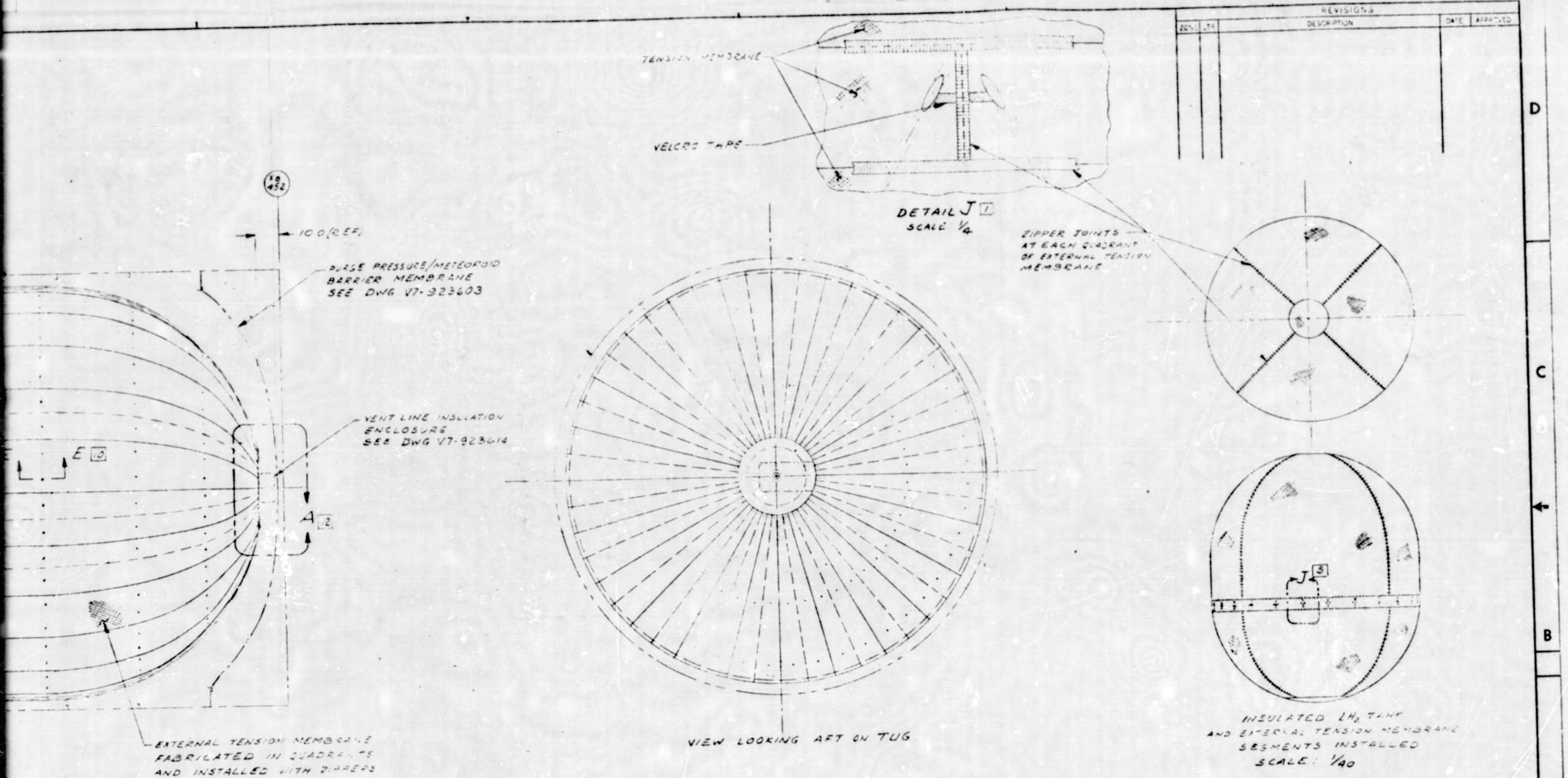


$\frac{1}{24}$ OF CIRCUMFERENCE
TYPICAL MLI MODULE
F 14
D 14
10
20
℄ TAB
10R
(TYP)
TYPICAL TAB ON EACH SHIELD. NUMBER AND LOCATION OF TABS TO BE DETERMINED ON MOCKUP
DETAIL F 16
SCALE: FULL SIZE
HOOK & PILE (VELCRO) ESTER, SEW TENSION ME SEW WITH (DACRON) PER FED STD 6 TO 8 STITCH
.80
3.25
.6 x 1.7 HOLE
CUT SIX RADIAL SLITS TO DIMENSIONS SHOWN
TYPICAL HOLE IN EACH MLI MODULE
SCALE: FULL SIZE
DETAIL D 17
THREADED INSERTS POTTED INTO HARDSPOT
1.00R
1.50
VIEW G-G 10
SCALE: ½ (MLI NOT
NOTE: FEWER SUPPORTS REQUIRED TO INSTAL WIRE MESH, THER SOME HARDSPOTS ARE SINGLE AND SOME DOUBLE INSERTS
SLITS IN ALTERNATE LAYERS OF MLI MODULES
VERTICAL SLITS IN EACH SHIELD OF SECOND, FOURTH, AND OUTSIDE MODULE
HORIZONTAL SLITS IN EACH SHIELD OF FIRST, THIRD, AND FIFTH MODULE
NOTE: INSULATION TO BE INSTALLED WITH STRUTS IN PLACE
VIEW OF LH_2 TANK SUPPORT STRUTS AND CUTOUTS IN INSULATION (EXTERNAL TENSION MEMBRANE NOT SHOWN IN THIS VIEW)
V7-923612
F-2
19
18
17
16
15
14
13


HOOK & PILE TAPE (VELCRO) 100% POLYESTER, SEWN TO THE TENSION MEMBRANE SEW WITH POLYESTER (DACRON) THREAD SIZE'E' PER FED. STD NO 751 6 TO 8 STITCHES PER INCH
MLI GORE SEGMENTS 24 REQUIRED
ACCESS DOOR MLI PANEL
PURGE GAS PASSAGES CUT INTO SUPPORT RING
LH2 TANK ACCESS DOOR V7-923608 (REF)
VELCRO TAPE
VENT LINE INSULATION ENCLOSURE (REF)
LOX TANK
C
C 22
TANK
DETAIL OF TANK ACCESS DOOR INSULATION
HALF SIZE
ADED-INSERTS POTTED HARDSPOT
.100R
.150
LH2 FILL & VENT LINE (FOR INSULATION SEE DWG V7-923614)
SUPPORT POST - FIBERGLASS/EPOXY .015 WALL THICKNESS
SINGLE ALUMINIZED KAPTON (SAK) EMBOSSED AND PERFORATED (2%) 30 SHEETS PER .50 INCH 6 MODULES MADE OF 5 SHEETS EACH
EXTERNAL TENSION MEMBRANE NOMEX MESH, TYPE HT-280-42, STERN & STERN TEXTILES, WITH ZIPPERS
H 12
FOR INSULATION OF LOX TANK SEE V7-923613
HARDSPOT ON DOOR
G-G 10
E 1/2 (MLI NOT SHOWN)
INTERNAL SUPPORT ALUMINUM WIRE MESH
G
G 13
1.0 GAP 1
NER SUPPORTS ARE UIRED TO INSTALL E MESH, THEREFORE, E HARDSPOTS ARE WITH GLE AND SOME WITH IBLE INSERTS 2
DETAIL A E
HALF SIZE
.100 DIA
.050 DIA 4 HOLES ON .625 DIA (FOR WRENCHING)
THREADED INSERT POTTED IN HARDSPOT PER PROCESS MA0606-039
HARDSPOT - FOAM FILLED HONEYCOMB PAD 2.0 INCH DIAMETER BONDED TO TANK WALL WITH LEFKOWELD 100 (MB0120-028) AND GLASS SCRIM CLOTH (MB0135-003) PER PROCESS MA0606-050
(442 PLACES)
VIEW E-E 6
TYPICAL INSULATION CROSS SECTION AND COMPONENTS
SCALE: FULL SIZE
.040 +.010 -.000
.940±.010
.520±.010
8-32 THREADS
.250 +.010 -.005 DIA
TYPICAL POST DETAIL H 10
SCALE: FULL SIZE
V7-923612
F-3
13 12 11 10 9 8 7



GORE SEGMENTS
EQUIRED
TENSION MEMBRANE
VELCRO TAPE
GE GAS PASSAGES
INTO SUPPORT RING
100 (REF)
PURGE PRESSURE/METEOROID
BARRIER MEMBRANE
SEE DWG V7-923603
LOX TANK
C
C 22
E
E 13
A 12
B 20
VENT LINE INSULATION
ENCLOSURE
SEE DWG V7-923614
NE INSULATION
SURE (REF)
SUPPORT POST - FIBERGLASS/EPOXY
.015 WALL THICKNESS
SINGLE ALUMINIZED KAPTON (SAK)
EMBOSSED AND PERFORATED (2%)
30 SHEETS PER .50 INCH
6 MODULES MADE OF 5 SHEETS EACH
FOR INSULATION
OF LOX TANK
SEE V7-923613
EXTERNAL TENSION MEMBRANE
FABRICATED IN QUADRANTS
AND INSTALLED WITH ZIPPERS
VIEW LOOKING AFT ON
1.0
GAP
1
G 13
HARDSPOT - FOAM FILLED HONEYCOMB
(442 PLACES) PAD 2.0 INCH DIAMETER
BONDED TO TANK WALL
WITH LEFKOWELD 109
(MB0120-028) AND GLASS
SCRIM CLOTH (MB0135-003)
PER PROCESS MA0606-050
ON
ENTS
CE
ORIENTATION VIEW
OF LH_2 TANK
SCALE: 1/20
6. TWIN INSERT HARDSPOT; ONE INSERT USED TO SUPPORT THE INTERNAL SUPPORT MEMBRANE (ALUMINUM WIRE MESH), THE SECOND INSERT SUPPORTS MLI POST.
GENERAL NOTES: 1. OPERATING TEMPERATURE(
THE MLI WILL BE EXPOSE
THE USE OF SINGLE ALUMI
2. THE EXTERNAL TENSION MEM
MESH WILL PREVENT MLI F
(ΔP = .01 PSIG).
2 3. THE INTERNAL SUPPORT MEM
WIRE MESH TO BE STRETCH-
NUMBER OF SEGMENTS TO BE BASED
4. TO PREVENT JOINT SEPA
TO MODULE WITH TABS US
TECHNIQUE. COAT KAPTON
FEP COATING LOCALLY AT
1 5. GAP PROVIDED BETWEE
MLI IS FOR CONDITIONI
SEE SCHEMATIC V7-923
AND VENTING SYSTEM.
V7-923612
F-4
9
8
7
6
5
4



GENERAL NOTES:

1. OPERATING TEMPERATURE (+200°F MAX) THAT THE MLI WILL BE EXPOSED TO DETERMINED THE USE OF SINGLE ALUMINIZED KAPTON.
2. THE EXTERNAL TENSION MEMBRANE - "NOMEX" MESH WILL PREVENT MLI FROM BALOONING (ΔP = .01 PSIG).
3. (2) THE INTERNAL SUPPORT MEMBRANE - ALUMINUM WIRE MESH TO BE STRETCH-FORMED TO CONTOUR. NUMBER OF SEGMENTS TO BE BASED ON MATERIAL AVAILABILITY
4. TO PREVENT JOINT SEPARATION, JOIN MODULE TO MODULE WITH TABS USING HEAT SEALING TECHNIQUE. COAT KAPTON TO KAPTON WITH FEP COATING LOCALLY AT JOINTS
5. (1) GAP PROVIDED BETWEEN THE TANK AND MLI IS FOR CONDITIONING OF INSULATION SEE SCHEMATIC V7-923615 FOR CONDITIONING AND VENTING SYSTEM.
6. TWIN INSERT HARDSPOT; ONE INSERT USED TO SUPPORT THE INTERNAL SUPPORT MEMBRANE (ALUMINUM WIRE MESH), THE SECOND INSERT SUPPORTS MLI POST.

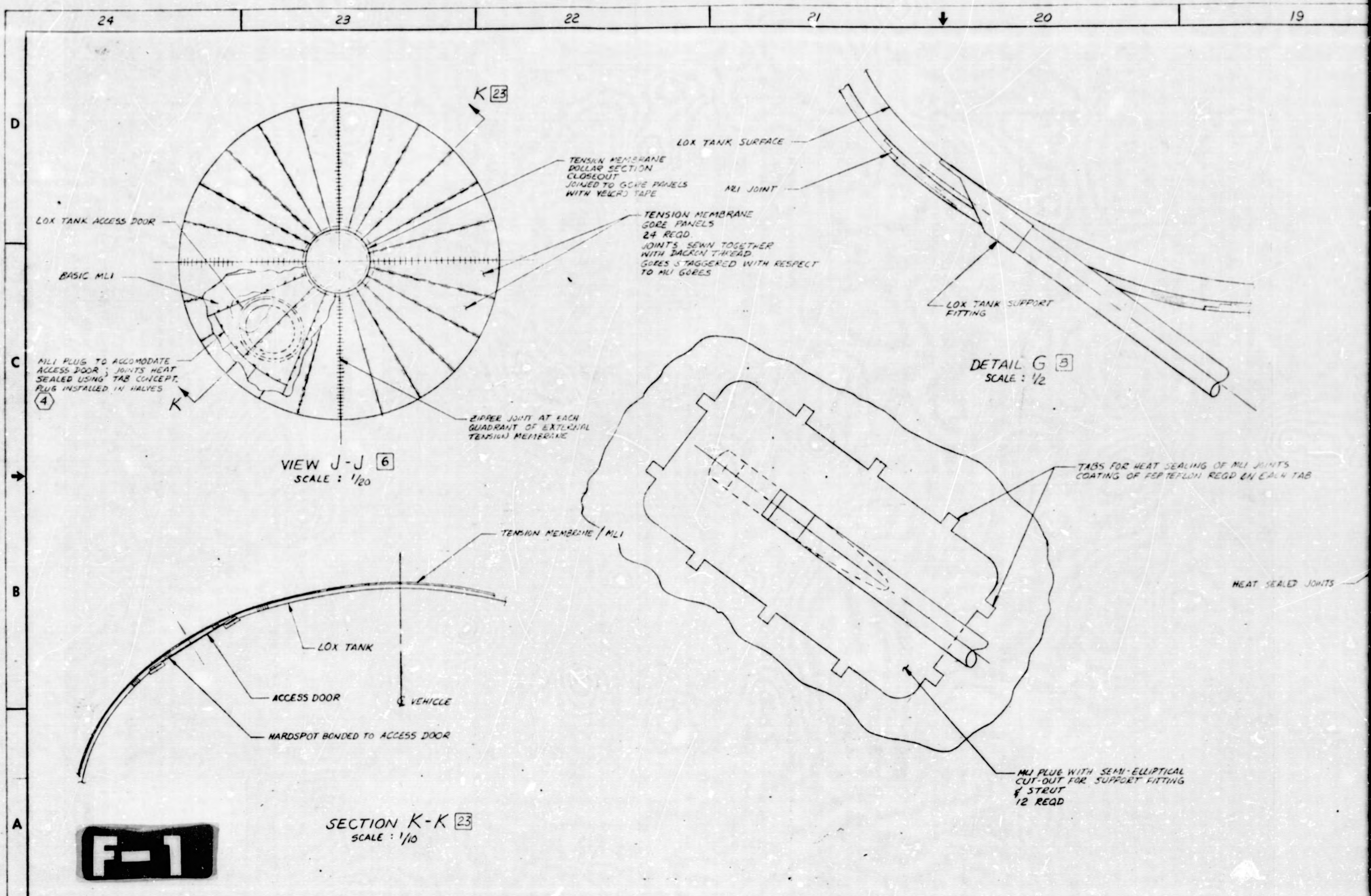
Figure 6.4-3
Layout-Insulation, Multi Layer Liquid Hydrogen Tank Space Tug

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	V7-923612

SCALE NOTED

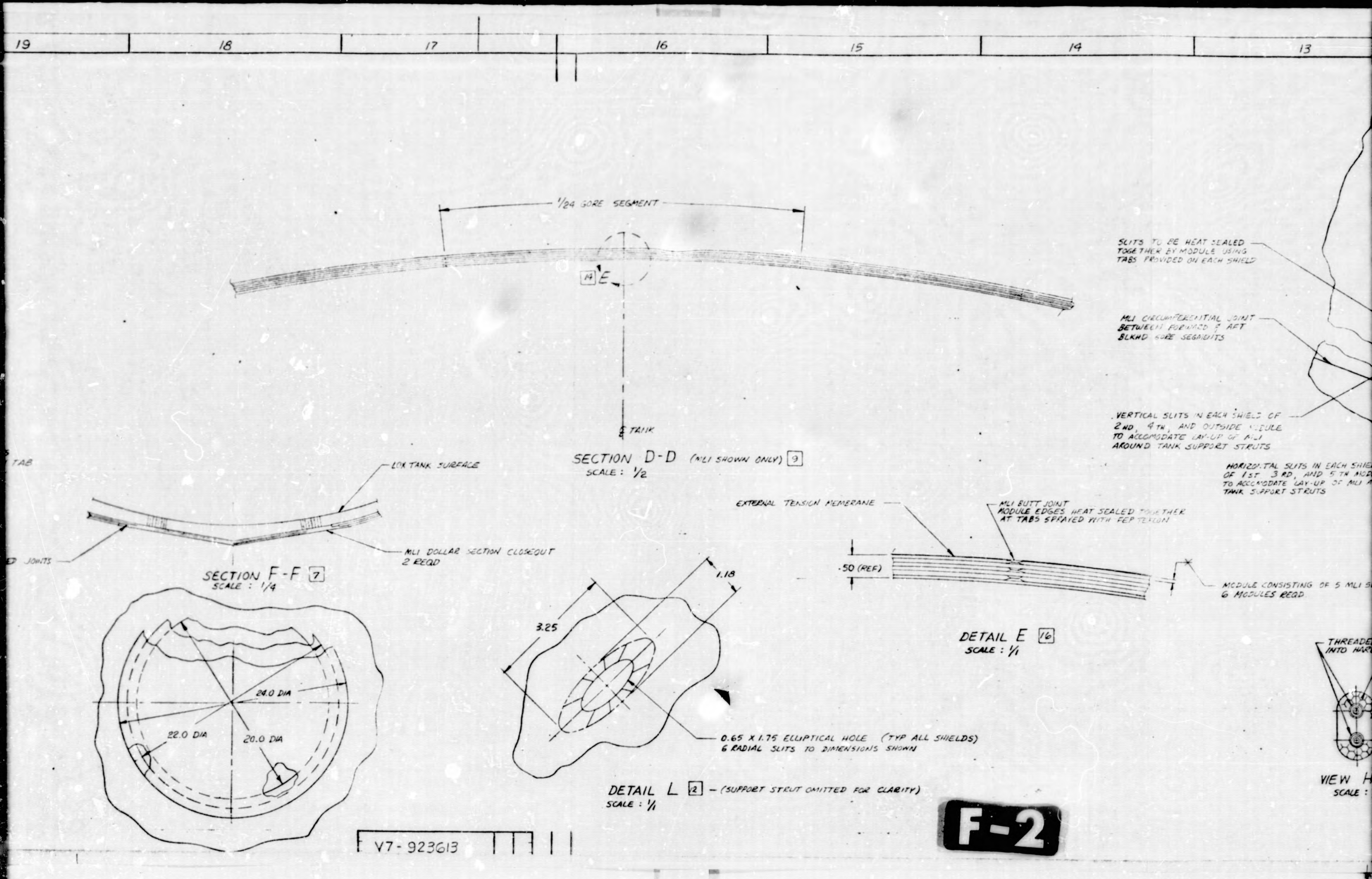
F-5

6 5 4 3 2 1

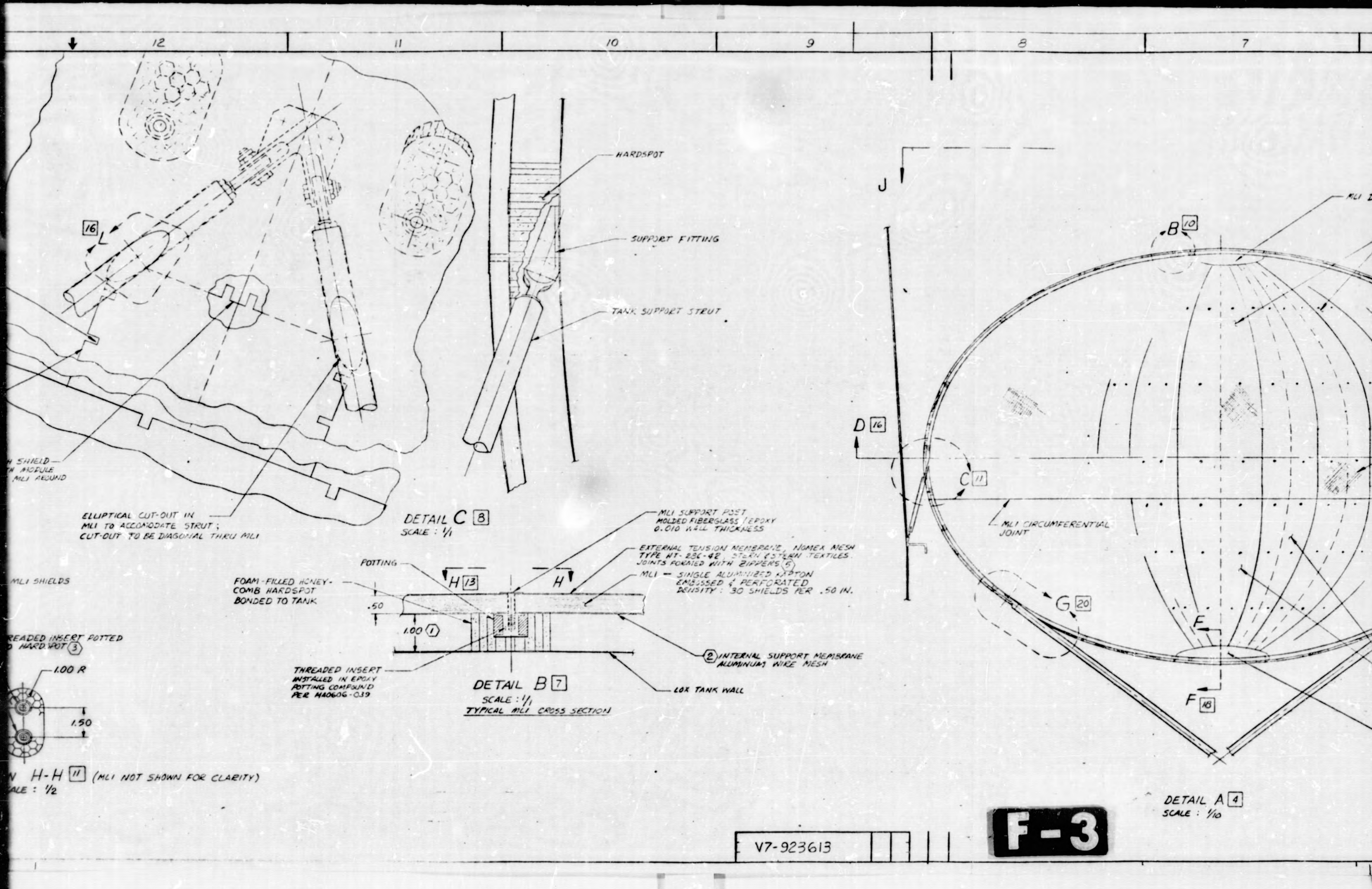


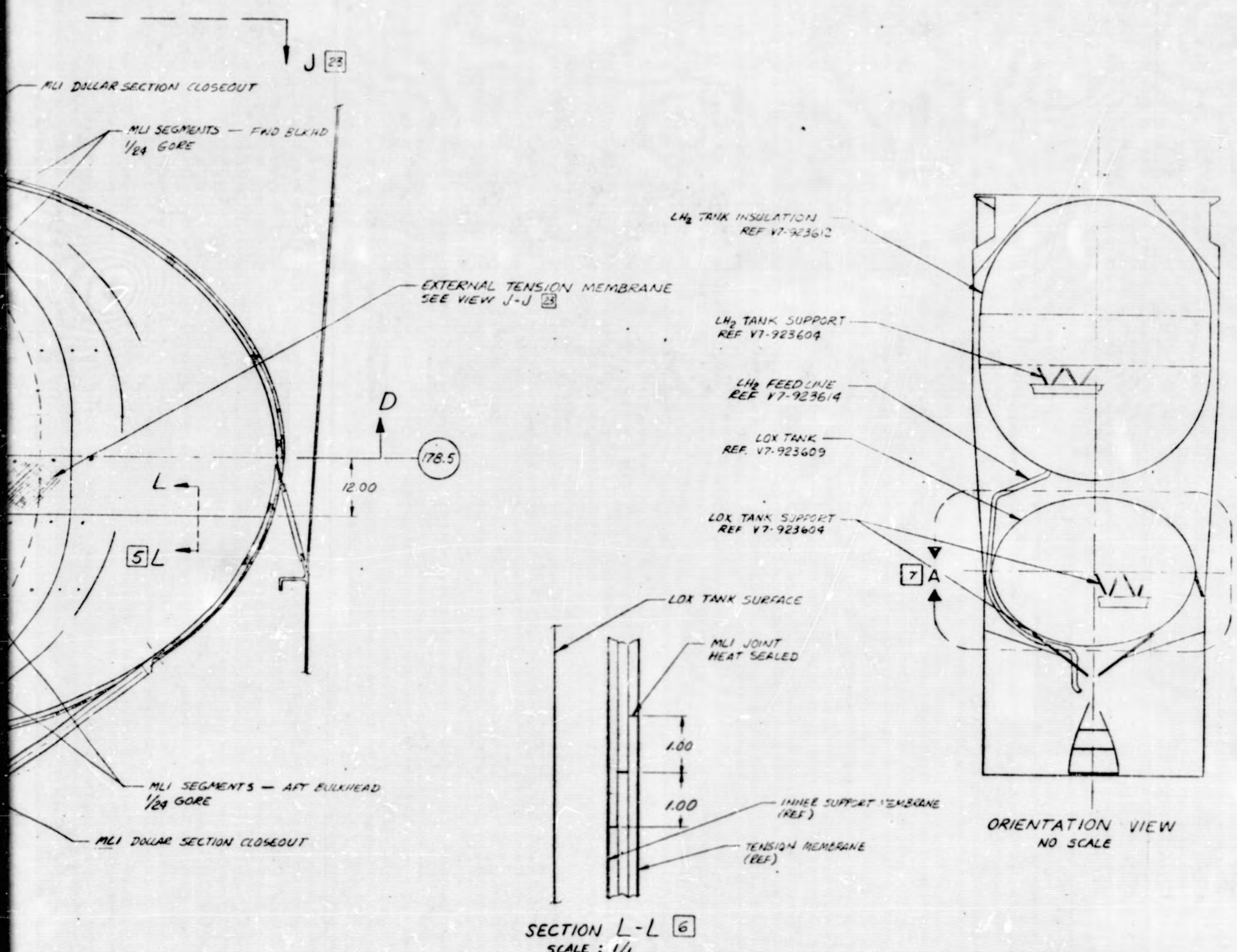
24
23
22
21
20
19
D
C
B
A
K 23
LOX TANK ACCESS DOOR
BASIC MLI
MLI PLUG TO ACCOMODATE ACCESS DOOR ; JOINTS HEAT SEALED USING TAB CONCEPT. PLUG INSTALLED IN HALVES
4
K
TENSION MEMBRANE DOLLAR SECTION CLOSEOUT JOINED TO GORE PANELS WITH VELCRO TAPE
TENSION MEMBRANE GORE PANELS 24 REQD. JOINTS SEWN TOGETHER WITH DACRON THREAD GORES STAGGERED WITH RESPECT TO MLI GORES
ZIPPER JOINT AT EACH QUADRANT OF EXTERNAL TENSION MEMBRANE
VIEW J-J 6
SCALE : 1/20
LOX TANK SURFACE
MLI JOINT
LOX TANK SUPPORT FITTING
DETAIL G 8
SCALE : 1/2
TABS FOR HEAT SEALING OF MLI JOINTS COATING OF FEP TEFLON REQD ON EACH TAB
HEAT SEALED JOINTS
MLI PLUG WITH SEMI-ELLIPTICAL CUT-OUT FOR SUPPORT FITTING & STRUT 12 REQD
TENSION MEMBRANE / MLI
LOX TANK
ACCESS DOOR
℄ VEHICLE
HARDSPOT BONDED TO ACCESS DOOR
SECTION K-K 23
SCALE : 1/10

F-1



19
18
17
16
15
14
13
1/24 GORE SEGMENT
14 E
℄ TANK
SECTION D-D (MLI SHOWN ONLY) 9
SCALE : 1/2
SLITS TO BE HEAT SEALED TOGETHER BY MODULE USING TABS PROVIDED ON EACH SHIELD
MLI CIRCUMFERENTIAL JOINT BETWEEN FORWARD & AFT BLKHD GORE SEGMENTS
VERTICAL SLITS IN EACH SHIELD OF 2ND, 4TH, AND OUTSIDE MODULE TO ACCOMODATE LAY-UP OF MLI AROUND TANK SUPPORT STRUTS
HORIZONTAL SLITS IN EACH SHIE OF 1ST 3RD, AND 5TH MOD TO ACCOMODATE LAY-UP OF MLI A TANK SUPPORT STRUTS
TAB
LOX TANK SURFACE
JOINTS
MLI DOLLAR SECTION CLOSEOUT 2 REQD
SECTION F-F 7
SCALE : 1/4
24.0 DIA
22.0 DIA
20.0 DIA
3.25
1.18
0.65 X 1.75 ELLIPTICAL HOLE (TYP ALL SHIELDS) 6 RADIAL SLITS TO DIMENSIONS SHOWN
DETAIL L 12 – (SUPPORT STRUT OMITTED FOR CLARITY)
SCALE : 1/1
EXTERNAL TENSION MEMBRANE
MLI BUTT JOINT MODULE EDGES HEAT SEALED TOGETHER AT TABS SPRAYED WITH FEP TEFLON
.50 (REF)
MODULE CONSISTING OF 5 MLI S 6 MODULES REQD
DETAIL E 16
SCALE : 1/1
THREADE INTO HAR
VIEW H
SCALE :
V7-923613
F-2


12
11
10
9
8
7
HARDSPOT
SUPPORT FITTING
TANK SUPPORT STRUT
16 L
ELLIPTICAL CUT-OUT IN MLI TO ACCOMODATE STRUT; CUT-OUT TO BE DIAGONAL THRU MLI
DETAIL C 8
SCALE: 1/1
MLI SUPPORT POST MOLDED FIBERGLASS / EPOXY 0.010 WALL THICKNESS
EXTERNAL TENSION MEMBRANE, NOMEX MESH TYPE HT-286-42, STERN & STERN TEXTILES JOINTS FORMED WITH ZIPPERS 5
MLI — SINGLE ALUMINIZED KAPTON EMBOSSED & PERFORATED DENSITY: 30 SHIELDS PER .50 IN.
POTTING
FOAM-FILLED HONEYCOMB HARDSPOT BONDED TO TANK
H 13
H
.50
1.00 1
2 INTERNAL SUPPORT MEMBRANE ALUMINUM WIRE MESH
THREADED INSERT INSTALLED IN EPOXY POTTING COMPOUND PER M40606-039
DETAIL B 7
SCALE: 1/1
TYPICAL MLI CROSS SECTION
LOX TANK WALL
MLI SHIELDS
1.00 R
1.50
H-H 11 (MLI NOT SHOWN FOR CLARITY)
J
B 10
D 16
C 11
MLI CIRCUMFERENTIAL JOINT
G 20
F
F 18
DETAIL A 4
SCALE: 1/10
V7-923613
F-3



NOTES:

1. (1) 1.00" HIGH PURGE ANNULUS REQD FOR UNIFORM PRESSURE DISTRIBUTION AT BACKSIDE OF MLI & FOR CLEARANCE PURPOSES. FOR PURGE SYSTEM REQMTS & DETAILS, SEE V7-923615 SCHEMATIC & V7-923616 PRESSURE TRANSFER SYSTEM DWG.
2. (2) INTERNAL SUPPORT MEMBRANE — ALUMINUM WIRE MESH — SHALL BE STRETCH-FORMED TO DESIRED CONTOUR. NUMBER OF SEGMENTS TO BE DETERMINED BY MATERIAL AVAILABILITY.
3. INSULATION CONCEPT AT FEED & VENT LINE PENETRATIONS SHOWN ON V7-923614.
4. (3) WHERE SUPPORT IS REQD FOR INTERNAL MEMBRANE, TWO INSERTS SHALL BE PROVIDED IN EACH HARD-SPOT : ONE INSERT FOR AN MLI SUPPORT POST, AND ONE INSERT FOR FASTENING THE INTERNAL MEMBRANE
5. (4) REMOVABLE SECTIONS ARE REQD IN EXTERNAL TENSION MEMBRANE & INTERNAL SUPPORT MEMBRANE AT MLI PLUG LOCATIONS & AT SUPPORT STRUT PENETRATIONS.
6. MAX OPERATING TEMPERATURE OF +200°F DICTATED USE OF KAPTON AS SUBSTRATE MATL FOR MLI.
7. (5) EXTERNAL TENSION MEMBRANE PREVENTS MLI FROM BALLOONING DURING REPRESSURIZATION WHEN $\Delta P = 0.01$ PSI. NOMEX SELECTED PRIMARILY FOR ITS HIGH TEMP CAPABILITY.

Figure 6.4-4
Layout - LOX Tank Insulation

REVISIONS				
ZONE	LTR	DESCRIPTION	DATE	APPROVED

LAYOUT — LOX TANK INSULATION

SCALE: NOTED
SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION
12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	V7-923613

SCALE NOTED SHEET

V7-923613

F-4

The tension membrane is fabricated from Nomex mesh (Stern and Stern Textiles, Type HT-286-42) and is installed over the MLI in 24 gore segments. The segments are sewn together into quadrants, which are joined with zippers for easy access to the MLI. The tension membrane is supported from the MLI support posts and is designed so as not to load the MLI when cryogens are on board. Cutouts are made in the Nomex mesh to accommodate MLI penetrations, such as tank support struts. Special sections of the tension membrane, including the dollar section closeouts and access door panels, are installed with velcro tape.

6.5.2 Propellant Transfer Line Insulation

Each of the propellant tanks has a feedline and a fill/vent line which require insulation (Ref. Dwg. V7-923614, Figure 6.4-5). Both the LH_2 and LOX feedlines are insulated with MLI from the tank to the engine heat block. The fill/vent lines are insulated from the tank to a point 24 inches outboard, which results in optimum system performance. In addition, the LH_2 fill/vent line is insulated with polyurethane foam where it penetrates the sealed purge barrier and is exposed to condensable gases. The MLI used on the propellant transfer lines is aluminized Kapton, identical to the material on the tanks and is one-half inch thick at all locations.

The MLI on the feedlines is installed in one-third circumference segments and heat sealed together in modules. The MLI is wrapped directly around the lines, which have no inner annulus. A single row of retention posts is used to restrict movement of the MLI and the external tension membrane. The tension membrane (Nomex mesh, Stern and Stern Textiles, Inc., Type HT-286-42) is installed over the MLI and closed with a zipper joint. Circumferential joints in the MLI are step joints and are heat sealed. Circumferential tension membrane joints are effected with Velcro tape.

The fill/vent line insulation is a special extension of the basic MLI on the propellant tanks. At these locations the aluminum mesh of the inner annulus is formed into a box, which encloses the fill/vent line. MLI is installed over the mesh in flat segments and retained by support posts. This design exposes the fill/vent line insulation to the same purge and preconditioning cycle as the tanks.

6.5.3 RCS Accumulator Insulation

Multi-layer insulation is installed directly over the GH_2 and GO_2 RCS accumulator tanks and transfer lines (Ref. Dwg. V7-923617, Figure 6.4-6). MLI modules for the accumulators are composed of pre-formed, semi-spherical segments. The modules are staggered by rotation, and the circumferential joints are heat sealed together at edge tabs. No MLI support posts are required with this configuration. Total MLI thickness is one-half inch. It should be noted, however, that pre-formed, embossed Kapton requires further development; if large pre-formed segments become economical, they could be used on the main propellant tanks.

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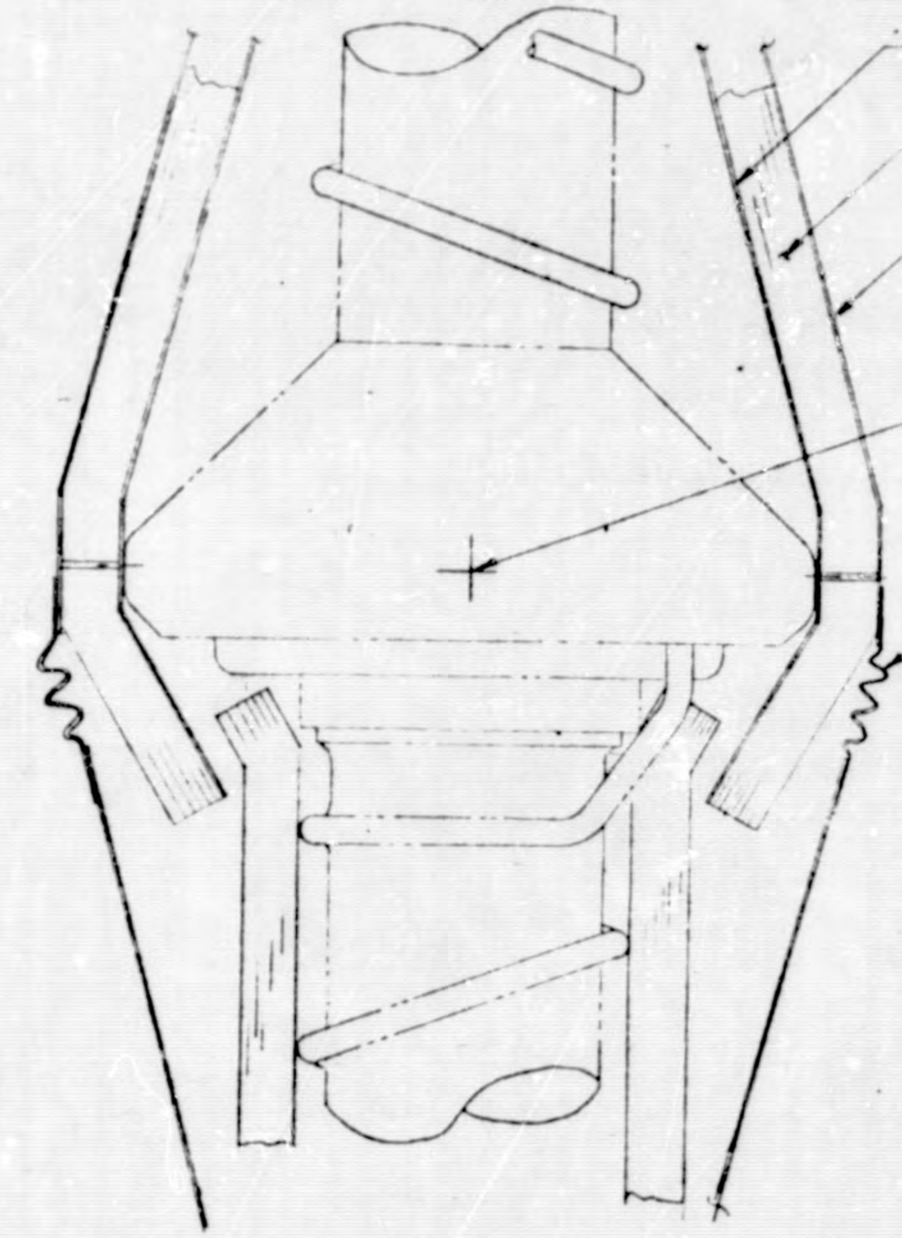
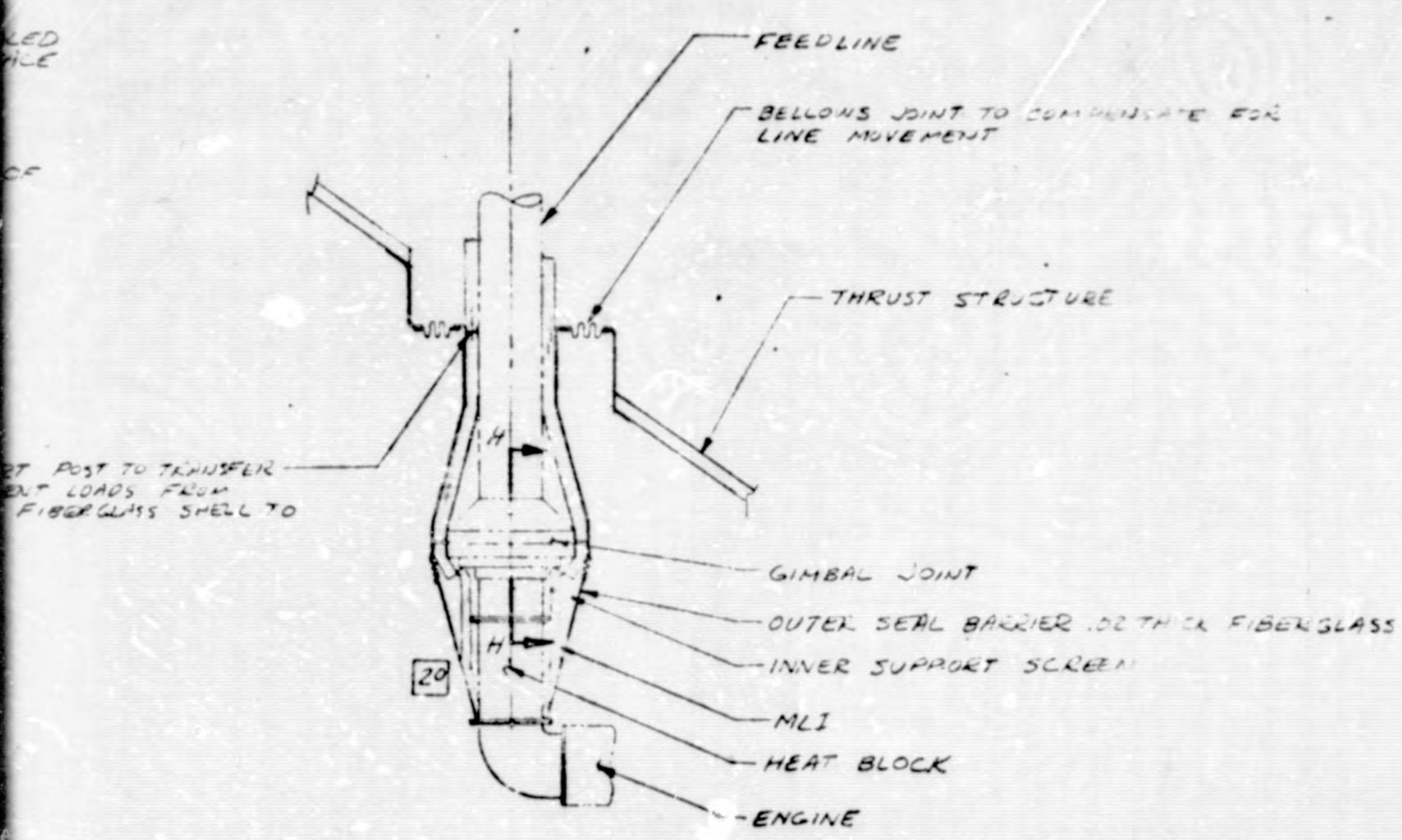


SUPPORT POST TO TR
MOVEMENT LOADS FR
OUTER FIBERGLASS S
LINE

SECTION H-H $\boxed{18}$
SCALE: FULL

TYPICAL INSULATION INSTL OVER GIMBAL JOINTS EXCEPT JOINTS INSIDE PURGED COMPARTMENT OF TUG. JOINTS IN COMPARTMENT HAVE NO OUTER FIBERGLASS COVER OR BELLOWS JOINT. A TENSION MEMBRANE WITH FLEX MESH IS INSTALLED INSTEAD.

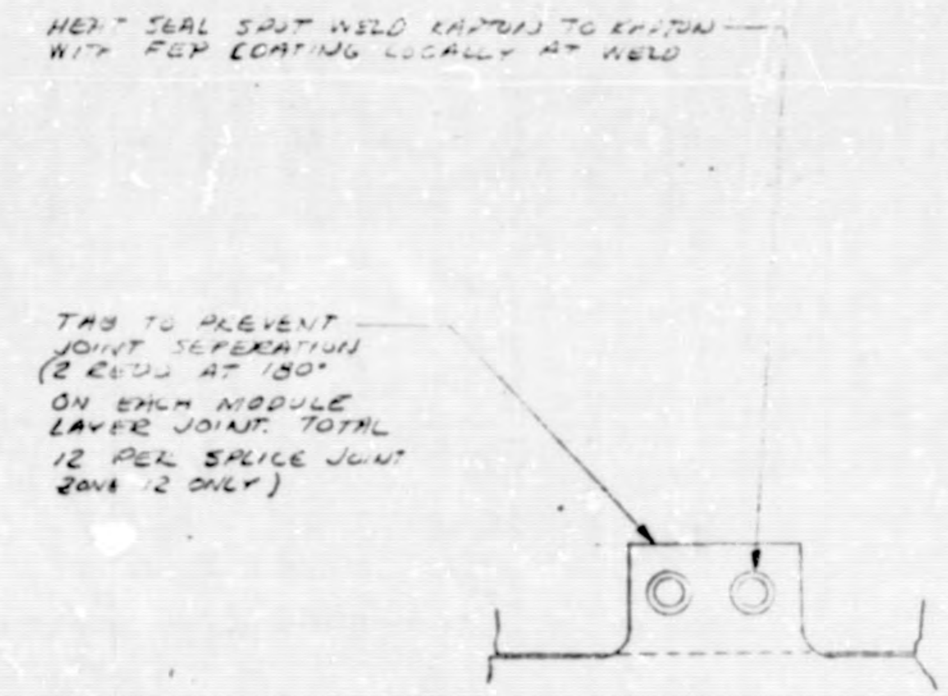
F-1



DETAIL G 3

SCALE: 1/4

TYPICAL LOX & LH_2 FEEDLINE ATTACHMENT TO ENGINE.



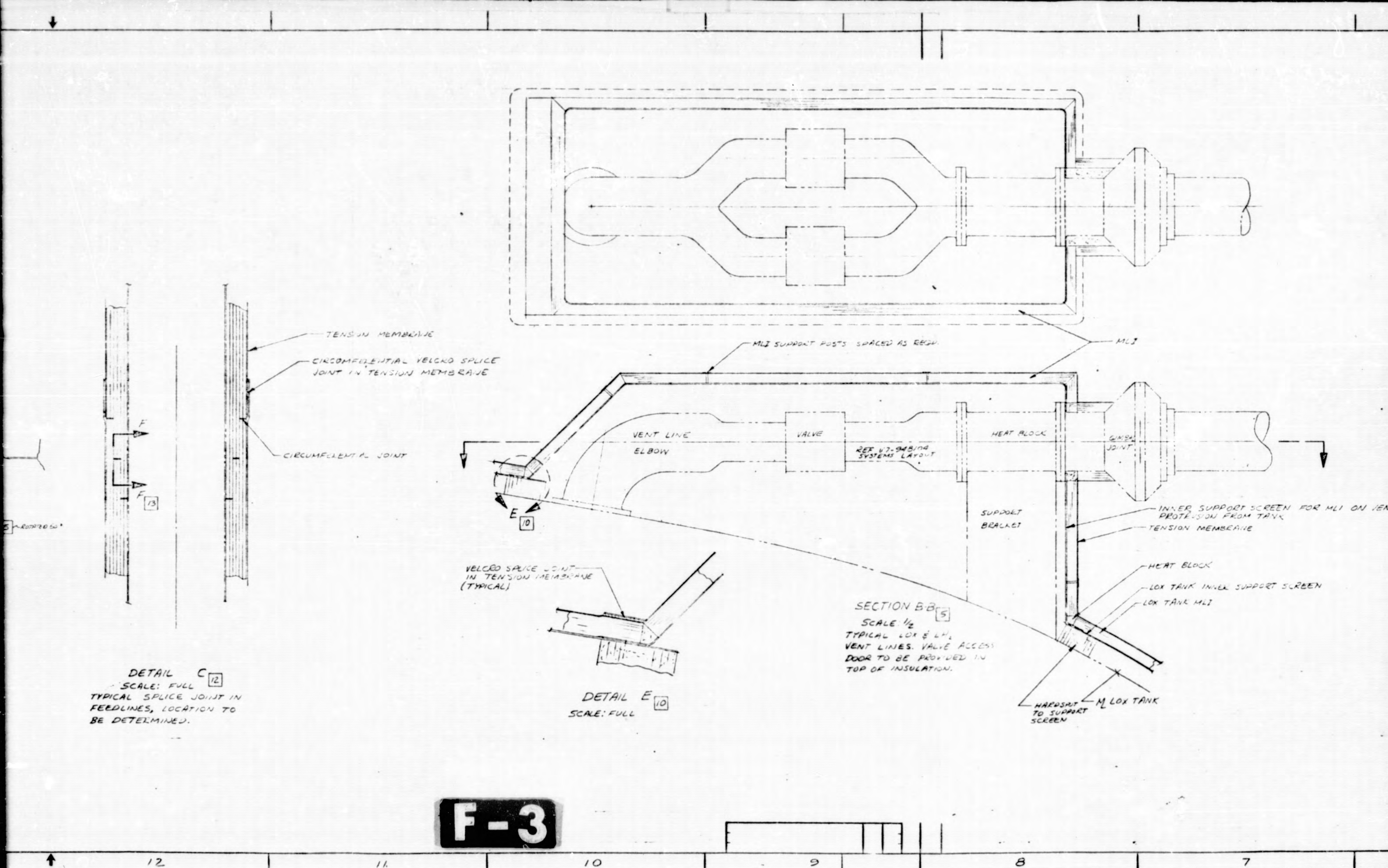
VIEW F-F

SCALE: FULL 12 5 ROTATED 60°

TYPICAL JOINT TAB.

F-2

18 17 16 15 14 13


TENSION MEMBRANE
CIRCUMFERENTIAL VELCRO SPLICE JOINT IN TENSION MEMBRANE
CIRCUMFERENTIAL JOINT
F
F 13
MLI SUPPORT POSTS SPACED AS REQD.
MLI
VENT LINE ELBOW
VALVE
REF V7-948174 SYSTEMS LAYOUT
HEAT BLOCK
GIMBAL JOINT
E 10
SUPPORT BRACKET
INNER SUPPORT SCREEN FOR MLI ON VENT PROTRUSION FROM TANK
TENSION MEMBRANE
HEAT BLOCK
LOX TANK INNER SUPPORT SCREEN
LOX TANK MLI
VELCRO SPLICE JOINT IN TENSION MEMBRANE (TYPICAL)
SECTION B-B 5
SCALE 1/2
TYPICAL LOX & LH2 VENT LINES. VALVE ACCESS DOOR TO BE PROVIDED IN TOP OF INSULATION.
HARDSET TO SUPPORT SCREEN
℄ LOX TANK
DETAIL C 12
SCALE: FULL
TYPICAL SPLICE JOINT IN FEEDLINES, LOCATION TO BE DETERMINED.
DETAIL E 10
SCALE: FULL
12
11
10
9
8
7

F-3

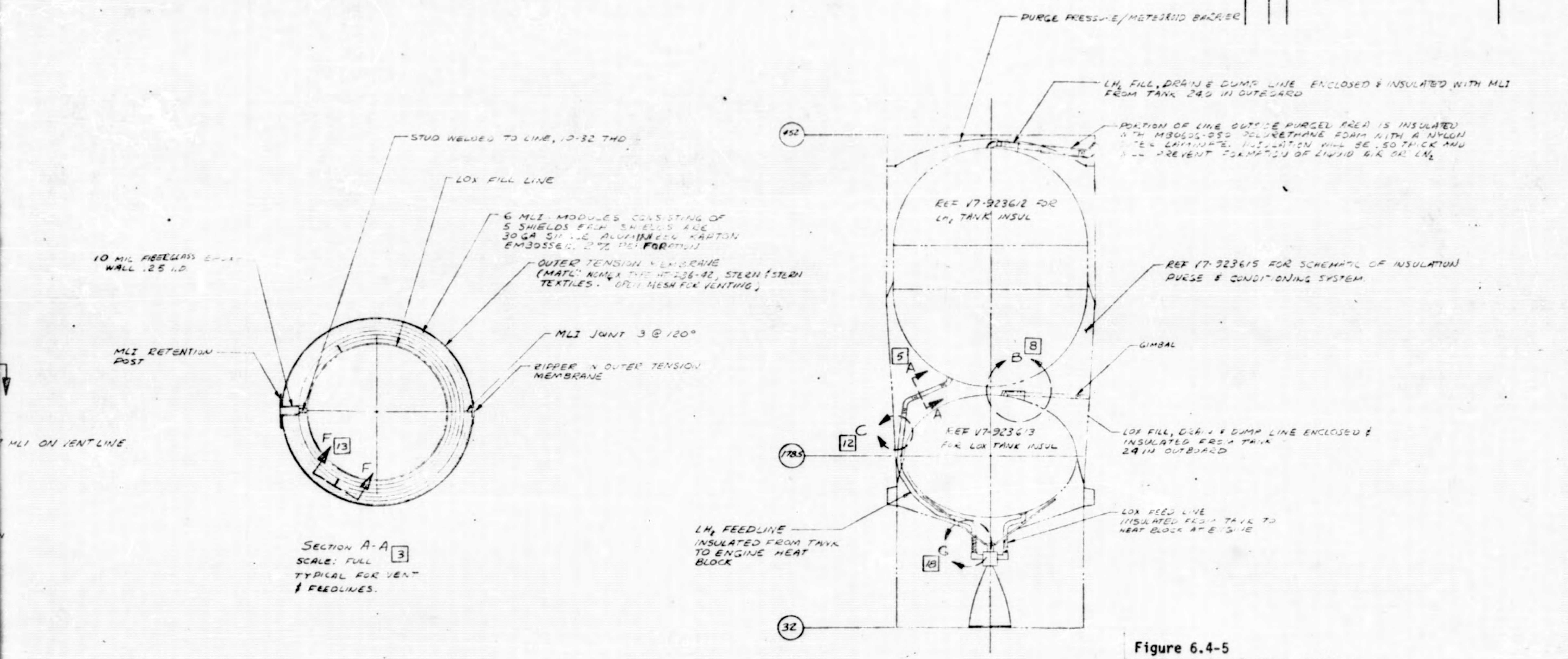


Figure 6.4-5
Layout-Insulation, Multi Layer, LH2 & LOX Fill & Feedlines Space Tug

GENERAL NOTES:

1. THE FEED AND VENT LINES WILL NOT HAVE PROVISIONS TO ALLOW A PURGE GAS FLOW OR A CONDITIONING (DRYING) GAS FLOW THRU THE MULTI LAYER INSULATION.
   RATIONALE:
   A. THE BASIC INSULATION PURGE SYSTEM WILL PREVENT INTRUSION OF MOISTURE OR CONTAMINATES.
   B. PRIOR TO LH_2 LOADING THE BASIC HELIUM PURGE SYSTEM WILL DIFFUSE A PORTION OF THE CONDENSIBLE GASES OUT OF THE FEED & VENT LINE INSUL.
   C. RESIDUAL CONDENSIBLE GASES IN THE LH_2 VENT LINE MAY BE CONDENSED BUT ARE OF SMALL VOLUME AND REPRESENT A SMALL AMOUNT OF OUT GASING AT ORBITAL ALTITUDE.
   D. SINCE THE FEEDLINE IS FORWARD DURING LAUNCH LH_2 WILL NOT BE IN THE LINE UNTIL ORBIT. HEAT LEAK & OUTGASING OF CONDENSED N_2 WILL BE NEGLIGIBLE.
2. RCS LINE NOT SHOWN. THE 3/8 DIA RCS LINE WILL BE INSULATED WITH MLI FROM THE TANK TO THE CHECK VALVE. REFERENCE MECHANICAL SYSTEMS SCHEMATIC
3. LINE SUPPORTS NOT SHOWN. SUPPORTS WILL BE NON-METALIC AND WILL NOT BE INSULATED.
4. GROUND VENT LINES (2.0 IN DIA) PER V7-941113 WILL BE INSULATED WITH 30 SHTS OF MLI (.50 THICK)

LAYOUT-INSULATION, MULTI LAYER, LH_2 & LOX FILL & FEEDLINES SPACE TUG

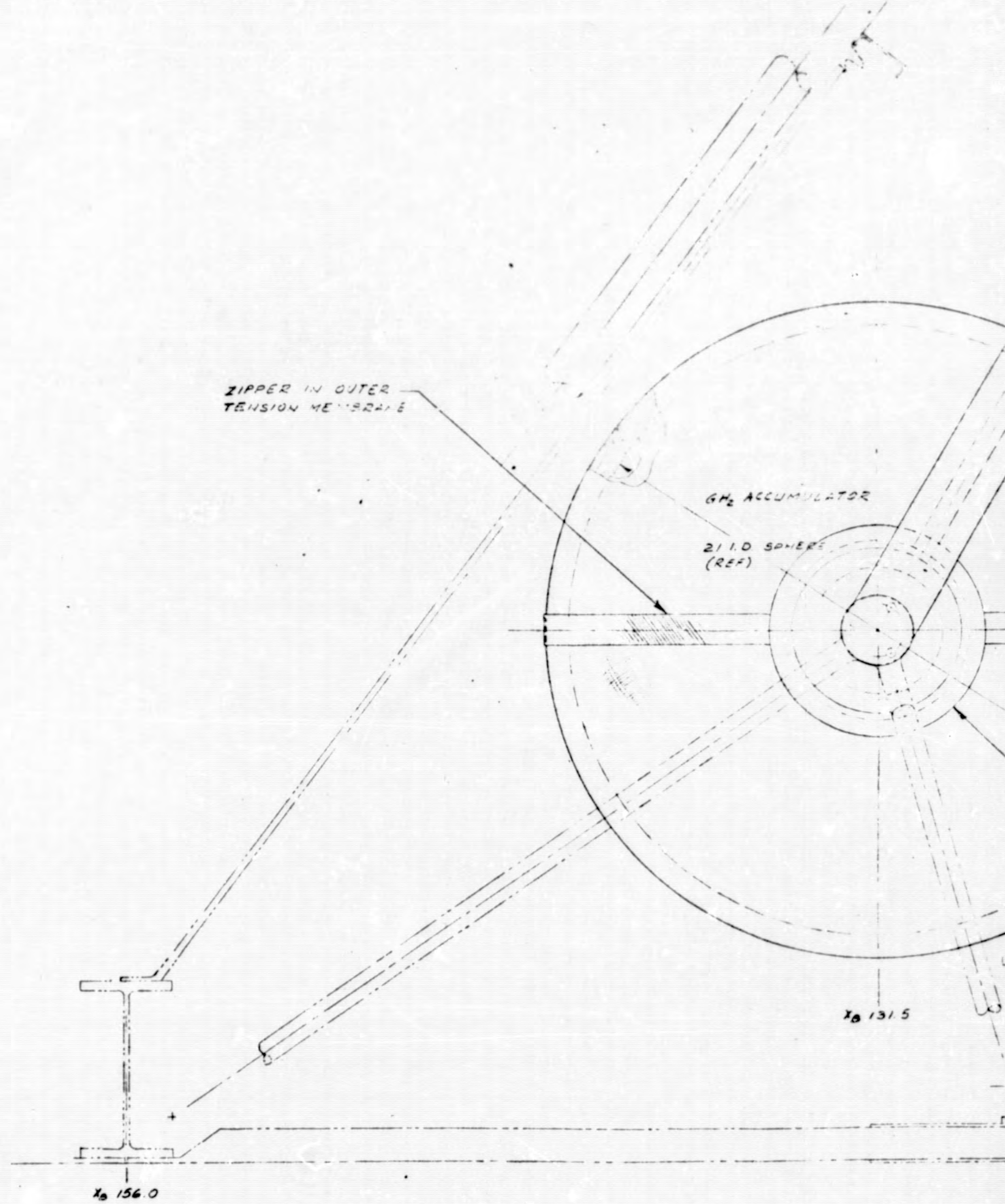
SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION
12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	V7-923614

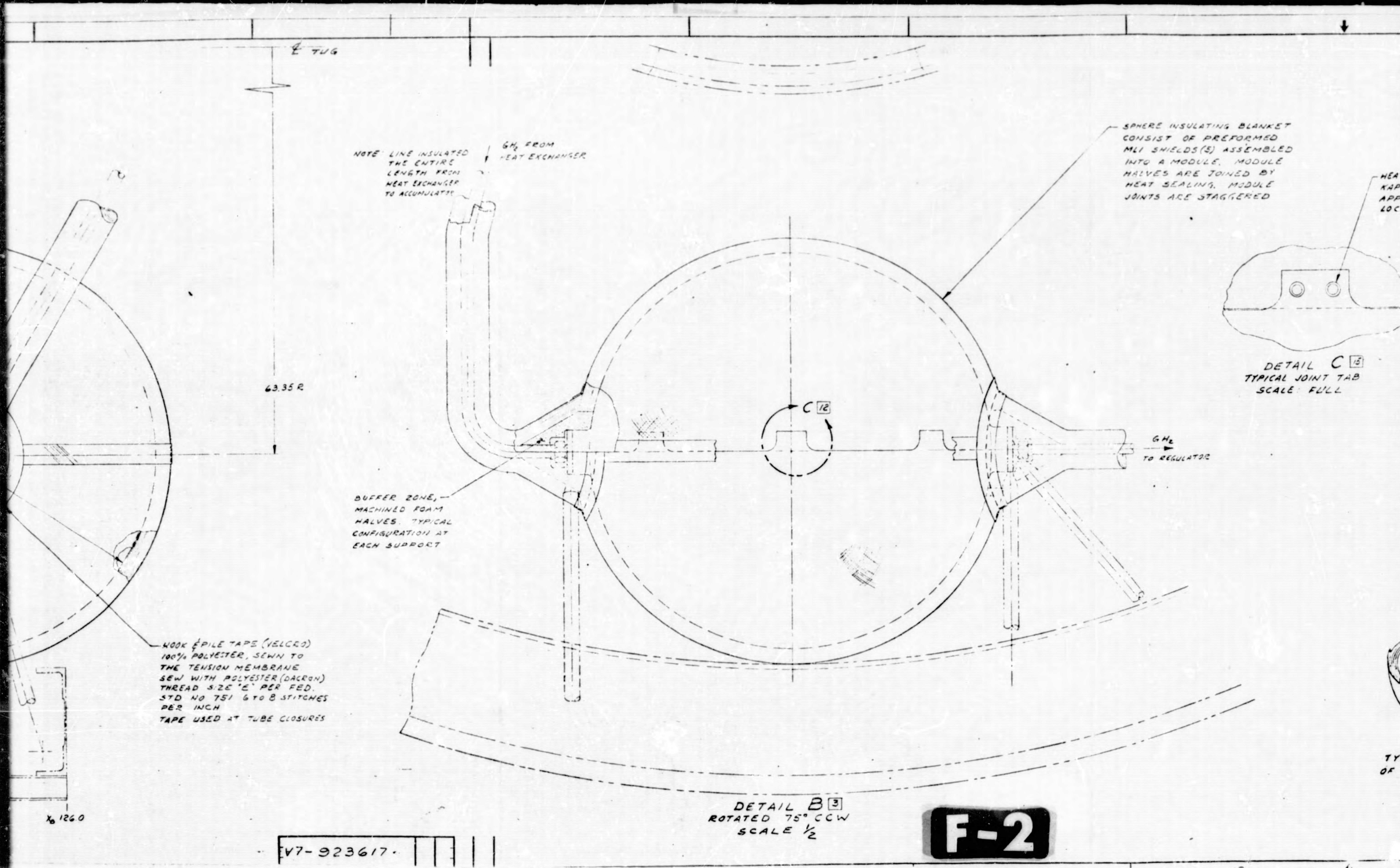
SCALE NOTED SHEET 1 OF 1

V7-923614

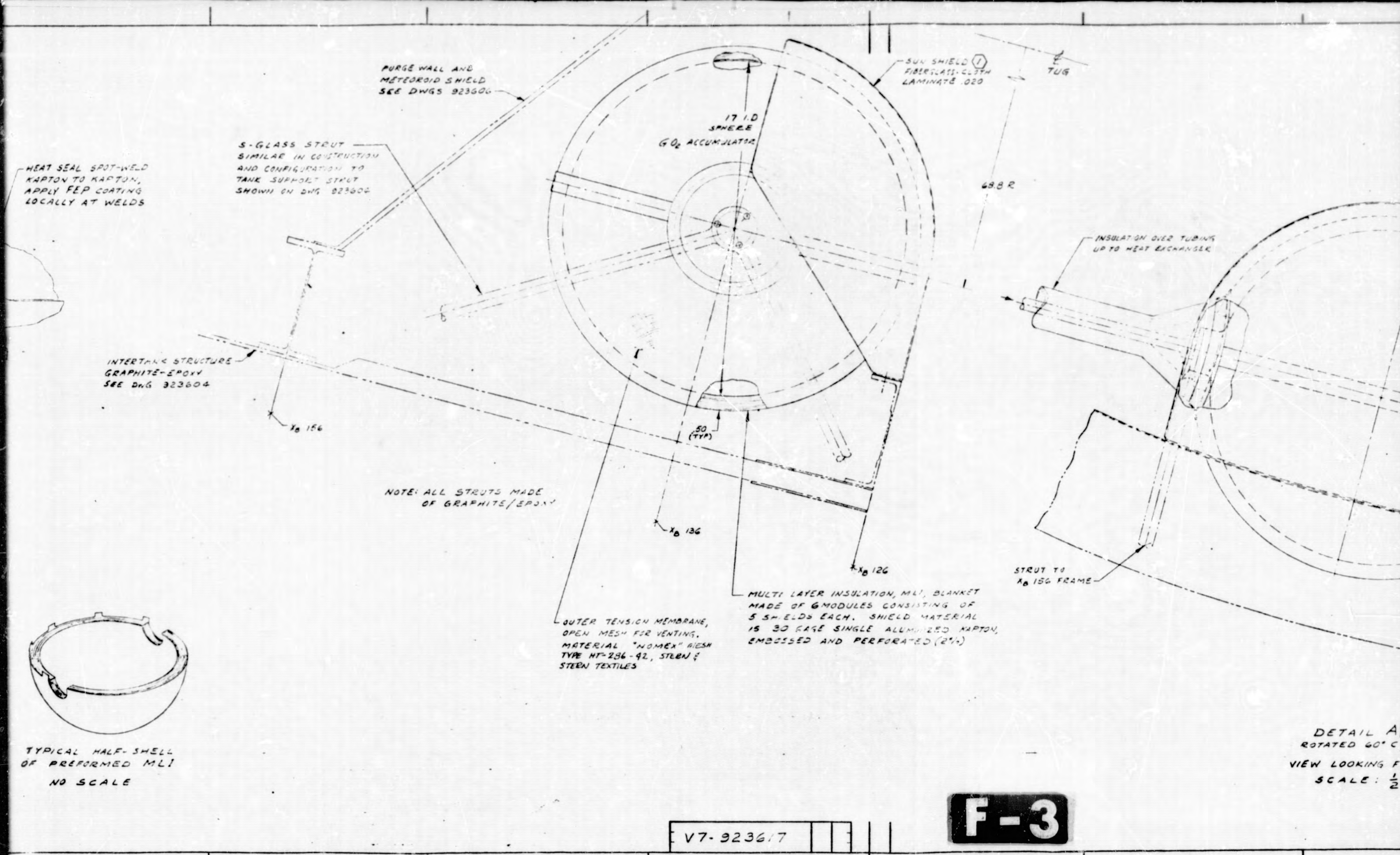
F-4


ZIPPER IN OUTER TENSION MEMBRANE
GH2 ACCUMULATOR
21 I.D SPHERE (REF)
XB 131.5
XB 156.0

F-1

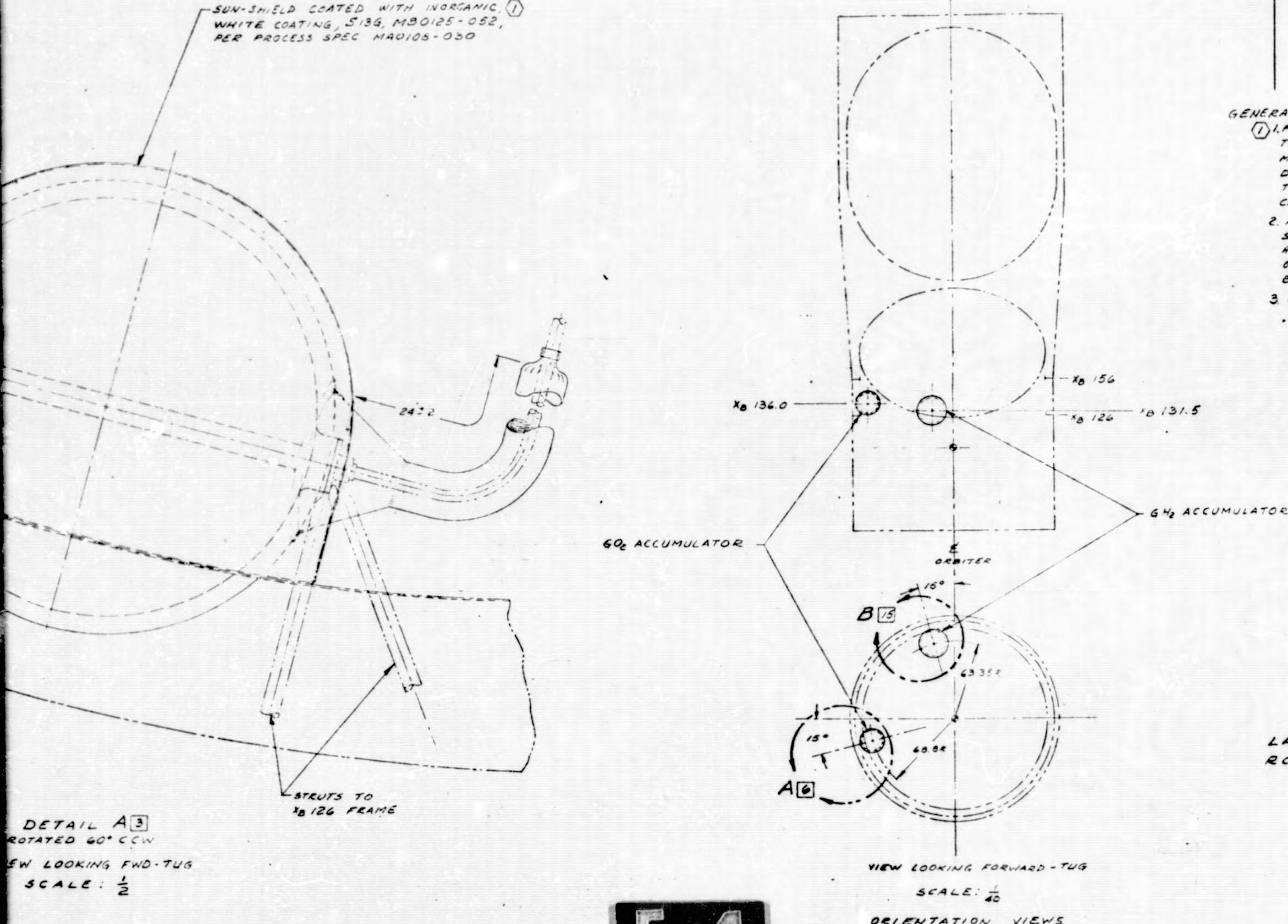

℄ TUG
63.35 R
NOTE: LINE INSULATED THE ENTIRE LENGTH FROM HEAT EXCHANGER TO ACCUMULATOR
GH_2 FROM HEAT EXCHANGER
SPHERE INSULATING BLANKET CONSIST OF PREFORMED MLI SHIELDS (5) ASSEMBLED INTO A MODULE. MODULE HALVES ARE JOINED BY HEAT SEALING. MODULE JOINTS ARE STAGGERED
DETAIL C 15
TYPICAL JOINT TAB
SCALE: FULL
C 12
GH_2
TO REGULATOR
BUFFER ZONE, MACHINED FOAM HALVES. TYPICAL CONFIGURATION AT EACH SUPPORT
HOOK & PILE TAPE (VELCRO) 100% POLYESTER, SEWN TO THE TENSION MEMBRANE
SEW WITH POLYESTER (DACRON) THREAD SIZE "E" PER FED. STD NO 751 6 TO 8 STITCHES PER INCH
TAPE USED AT TUBE CLOSURES
X_o 1260
DETAIL B 3
ROTATED 75° CCW
SCALE 1/2
V7-923617

F-2



PURGE WALL AND METEOROID SHIELD SEE DWGS 923606
SUN SHIELD (1) FIBERGLASS-CLOTH LAMINATE .020
℄ TUG
17 I.D SPHERE
GO_2 ACCUMULATOR
S-GLASS STRUT SIMILAR IN CONSTRUCTION AND CONFIGURATION TO TANK SUPPORT STRUT SHOWN ON DWG 923606
HEAT SEAL SPOT-WELD KAPTON TO KAPTON, APPLY FEP COATING LOCALLY AT WELDS
68.8 R
INSULATION OVER TUBING UP TO HEAT EXCHANGER
INTERTANK STRUCTURE GRAPHITE-EPOXY SEE DWG 923604
X_B 156
.50 (TYP)
NOTE: ALL STRUTS MADE OF GRAPHITE/EPOXY
X_B 136
X_B 126
STRUT TO X_B 156 FRAME
MULTI LAYER INSULATION, MLI, BLANKET MADE OF 6 MODULES CONSISTING OF 5 SHIELDS EACH. SHIELD MATERIAL IS 30 GAGE SINGLE ALUMINIZED KAPTON, EMBOSSED AND PERFORATED (2%)
OUTER TENSION MEMBRANE, OPEN MESH FOR VENTING. MATERIAL "NOMEX" MESH TYPE HT-286-42, STERN & STERN TEXTILES
TYPICAL HALF-SHELL OF PREFORMED MLI
NO SCALE
DETAIL A
ROTATED 60° C
VIEW LOOKING F
SCALE: 1/2
V7-923617
12
11
10
9
8
7
F-3

ZONE	LTR	DESCRIPTION	DATE	APPROVED



GENERAL NOTES

(1) 1. PRELIMINARY THERMAL ANALYSIS INDICATE THAT A SUN-SHIELD OVER EACH ACCUMULATOR MAY BE DESIRABLE. FURTHER ANALYSIS WILL DEFINE EXACT REQUIREMENTS, AND AT THAT TIME THE SUN-SHIELD CONFIGURATION MAY CHANGE FROM THE ONE SHOWN (ZONE 6).

2. ACCUMULATORS AND LINES LOCATED IN THE AFT SKIRT AREA ARE NOT PROTECTED BY THE TUG AIR ISOLATION (PURGE) SYSTEM DURING CRYOGENIC OPERATIONS, THE COMPONENTS ARE PROTECTED BY THE DRY N_2 PURGE IN THE ORBITER COMPARTMENT

3. OPERATING TEMPERATURES (°F):

	GH_2		GO_2	
	MAX	MIN	MAX	MIN
MLI EXTERNAL SURFACE	100	-300	100	-300
ACCUMULATOR SURFACE	-210	-235	-10	-55

LAYOUT-INSULATION, MULTILAYER
RCS ACCUMULATOR TANKS

Figure 6.4-6
Layout-Insulation, Multi Layer RCS Accumulator Tanks

F-4

6 - 29, 6 - 30

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	V7-923617

SCALE NOTED SHEET 1 of 1

The tension membrane (Nomex mesh) is installed over the MLI in halves and zippered together.

The inlet line to each accumulator is insulated over its entire length from the heat exchanger. The outlet line is insulated out to 24 inches from the accumulator to optimize system performance. In both cases the MLI consists of a one-half inch thick wrapping covered with a tension membrane. The membranes on the transfer lines are joined to the accumulator tension membrane with Velcro tape.

A solar radiation shield for the accumulators is shown on V7-923617 to indicate a conservative alternate design. Further analysis, however, is needed before the requirement for such a shield is firmly established. The shield will be deleted, if possible, for weight reduction.

6.5.4 Purge/Conditioning System

The purge/conditioning system consists of distribution manifolds under the MLI on the propellant tanks, a compartment diffuser, and pneumatic supply lines (Ref. Dwg. V7-923615, Figure 6.4-7 and 6.4-8). An umbilical connects the main pneumatic supply line to the gas supply bottles. The gas flows to a purge system control package inside the sealed structure, from which the gas can be directed to either the diffuser or the distribution manifolds.

The distribution manifolds are made from aluminum extrusion and are semi-circular in cross section with tabs provided for bonding to the tank surface. The manifolds completely encircle the girth of each propellant tank. Two rows of 0.125 inch diameter orifices are drilled in each manifold to distribute the purge gas evenly.

6.6 CRYOGENIC INSULATION MATERIALS DISCUSSION

6.6.1 Multiple Layer Insulation Film Selection

Research studies have been conducted by NR in an effort to optimize the thermal insulation configuration used to insulate cryogenic liquid tanks. The conclusion reached in these studies is that the most efficient cryogenic insulation configuration, per unit weight, consists of multiple layers of thin plastic film which is metallized on one surface with a highly reflective coating and which is embossed to minimize the areas of physical surface contact between adjacent film layers.

The original cryogenic insulation configuration developed by NR utilized polyester (Mylar) film which was embossed with a random "cut-bead" pattern and metallized on one surface. However, tests reveal that this polyester film will lose embossment height if it is heated above 140 F while under low compressive loads such as may occur during dry-gas purging of the insulation configuration. In the case of the proposed Space Tug the Cryogenic insulation configuration is expected to be heated to 200 F under load. Therefore, a change is necessary in the type of plastic film used for the insulation.

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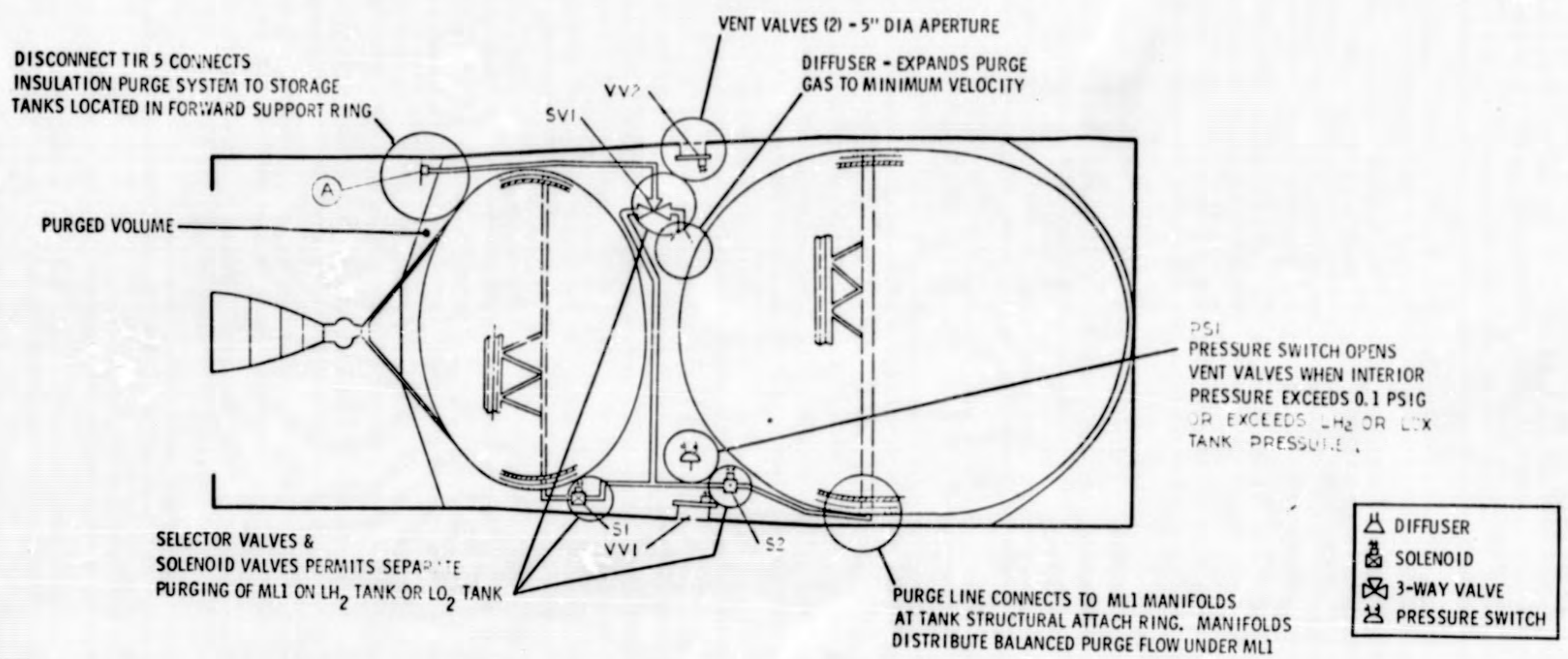
REVISIONS				
ZONE	LTR	DESCRIPTION	DATE	APPROVED

INSULATION PURGE SYSTEM



EVENT CONDITION	GROUND OPERATIONS			PRELAUNCH		LAUNCH	ORBIT	RE-ENTRY
	① STORAGE	PRECONDITION	② PURGE	AMBIENT	CRYO	CRYO-VENT	CRYO-SPACE	CRYO-REPRESS
SOURCE OF GAS	GSE	GSE	GSE	GSE	GSE	RESIDUAL	INTERNAL VOLATILES & LEAKS	FLIGHT TANK
GAS	N_2/DRY AIR	N_2	N_2	He	He	He		He
CONNECTOR TIR5	CONNECTED	CONNECTED	CONNECTED	CONNECTED	CONNECTED	CONNECTED	DISCONNECTED	CONNECTOR
SELECTOR VALVE SV 1	DIFFUSER	MLI MANIFOLDS	DIFFUSER	MLI MANIFOLD	DIFFUSER	DIFFUSER	DIFFUSER	DIFFUSER
SOLENOID S1	CLOSED	OPEN/CLOSED	CLOSED	OPEN	CLOSED	CLOSED	CLOSED	CLOSED
SOLENOID S2	CLOSED	CLOSED/OPEN	CLOSED	OPEN	CLOSED	CLOSED	CLOSED	CLOSED
VENT VALVE VV1	CLOSED	OPEN	CLOSED	OPEN	CLOSED	OPEN	OPEN	CLOSED
VENT VALVE VV2	CLOSED	OPEN	CLOSED	OPEN	CLOSED	OPEN	OPEN	CLOSED
FLOW PATH	TUG INTERIOR	THRU MLI	TUG INTERIOR	THRU MLI	TUG INTERIOR	VENT INTERIOR	VENT INTERIOR	TUG INTERIOR
FLOW RATE	MAKE-UP TO MAINTAIN 0.1 PSI	UP TO 400 SCFM	MAKE-UP TO MAINTAIN 0.1 PSI	10 SCFM	MAKE-up UP TO MAINTAIN 0.1PSI			0.1 PSI ABOVE FLIGHT AMBIENT
PRESSURE SWITCH PS 1	OPERATIVE	OPERATIVE	OPERATIVE	OPERATIVE	OPERATIVE	OPERATIVE	INERT	OPERATIVE
PURPOSE	ISOLATION FROM MOISTURE & CONTAMINATES	REMOVE MOISTURE ACCIDENTALLY INTRUDING INTO MLI	ISOLATION FROM MOISTURE & CONTAMINATES	FLUSH CONDENSABLE GAS FROM MLI	ISOLATION FROM CONDENSABLE GAS	EVACUATE He PURGE TO PRECLUDE OVERPRESSURIZATION & ENSURE EARLY THERMAL PROTECTION	EVACUATE MATL OUTGASS PRODUCTS, PROPELLANT & GAS LEAKAGE	REPRESSURIZE MLI & TUG TO RE-ENTRY PROFILE

① PURGE NOT REQUIRED IF TUG IS IN CONTROLLED ENVIRONMENT
② PURGE NOT REQUIRED WHILE ORBITER COMPARTMENT IS PURGED

6 - 33, 6 - 34

Figure 6.4-7

Schematic-Conditioning & Purge System, Multilayer Insulation, Tug, MCR 7007

1. SEE MECHANICAL SYSTEMS SCHEMATIC V7-94.B10 FOR He/N_2 SUPPLY (A).
GENERAL NOTES:

UNLESS OTHERWISE SPECIFIED: DIMENSIONS ARE IN INCHES
TOLERANCES ON:

DECIMALS	ANGLES
.XX= ± .03	
.XXX= ± .010	± 30'

HOLES NOTED "DRILL"

.013	THRU	.040:	+ .001 − .001
.041	THRU	.130:	+ .002 − .001
.131	THRU	.229:	+ .003 − .001
230	THRU	.500:	+ .004 − .001
.501	THRU	.750:	+ .005 − .001
.751	THRU	1.000:	+ .007 − .001
1.001	THRU	2.000:	+ .010 − .001

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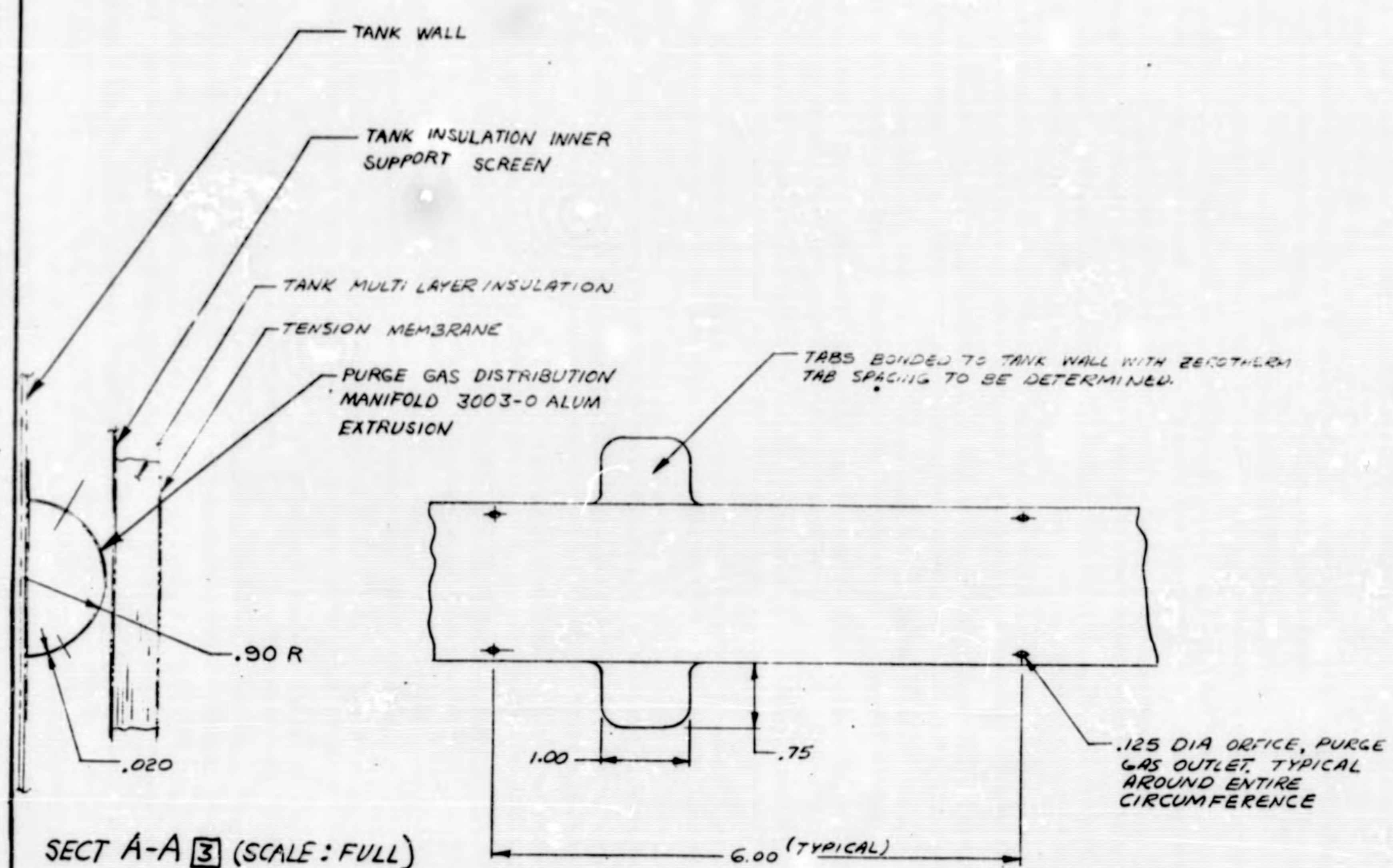
SCHEMATIC - CONDITIONING & PURGE SYSTEM, MULTI LAYER INSULATION, TUG, MCR 7007

SIZE D | CODE IDENT NO. 03953 | DRAWING NO. V7-923615

SCALE NONE | SHEET

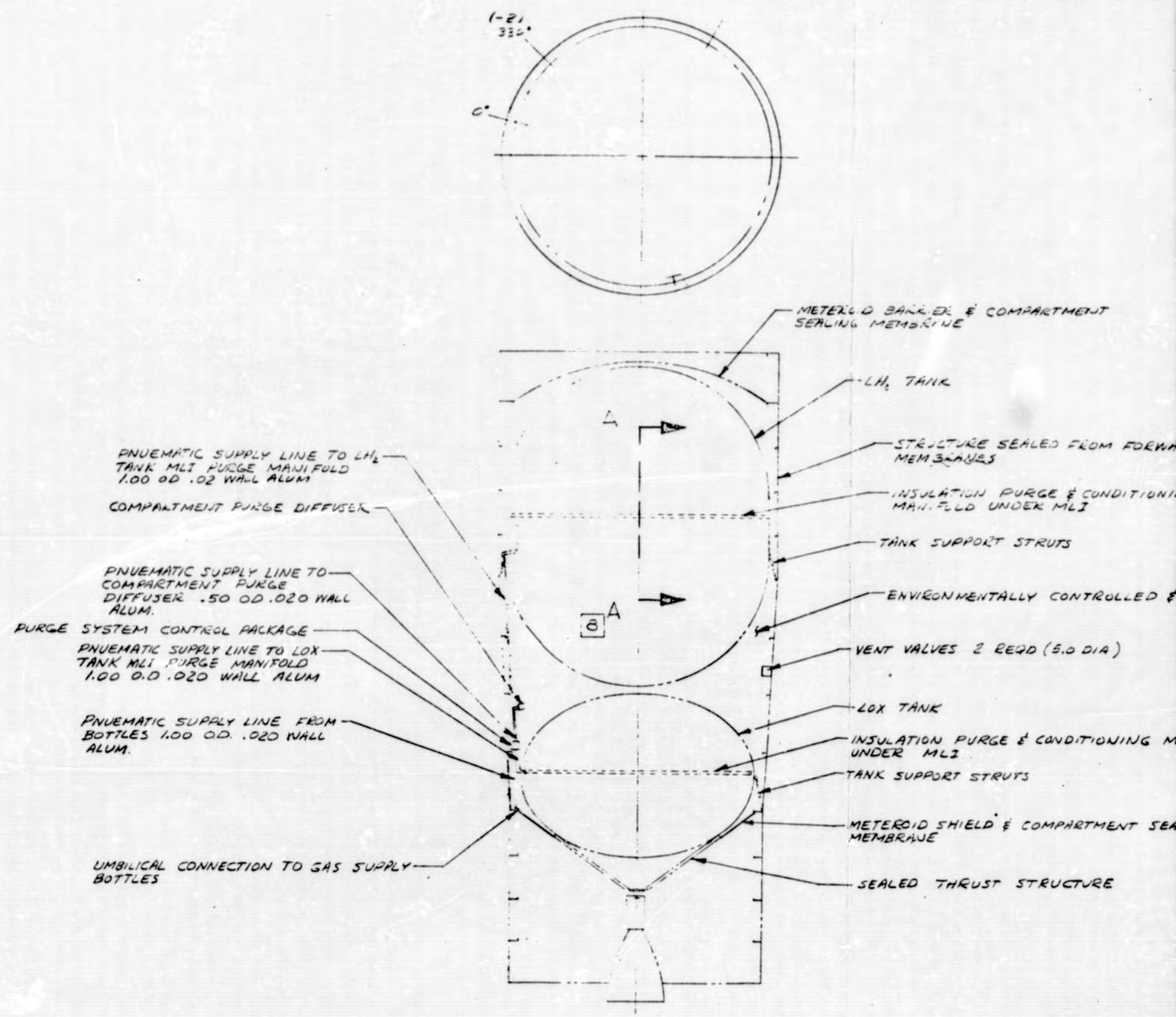
V7-923615



SECT A-A $\boxed{3}$ (SCALE : FULL)

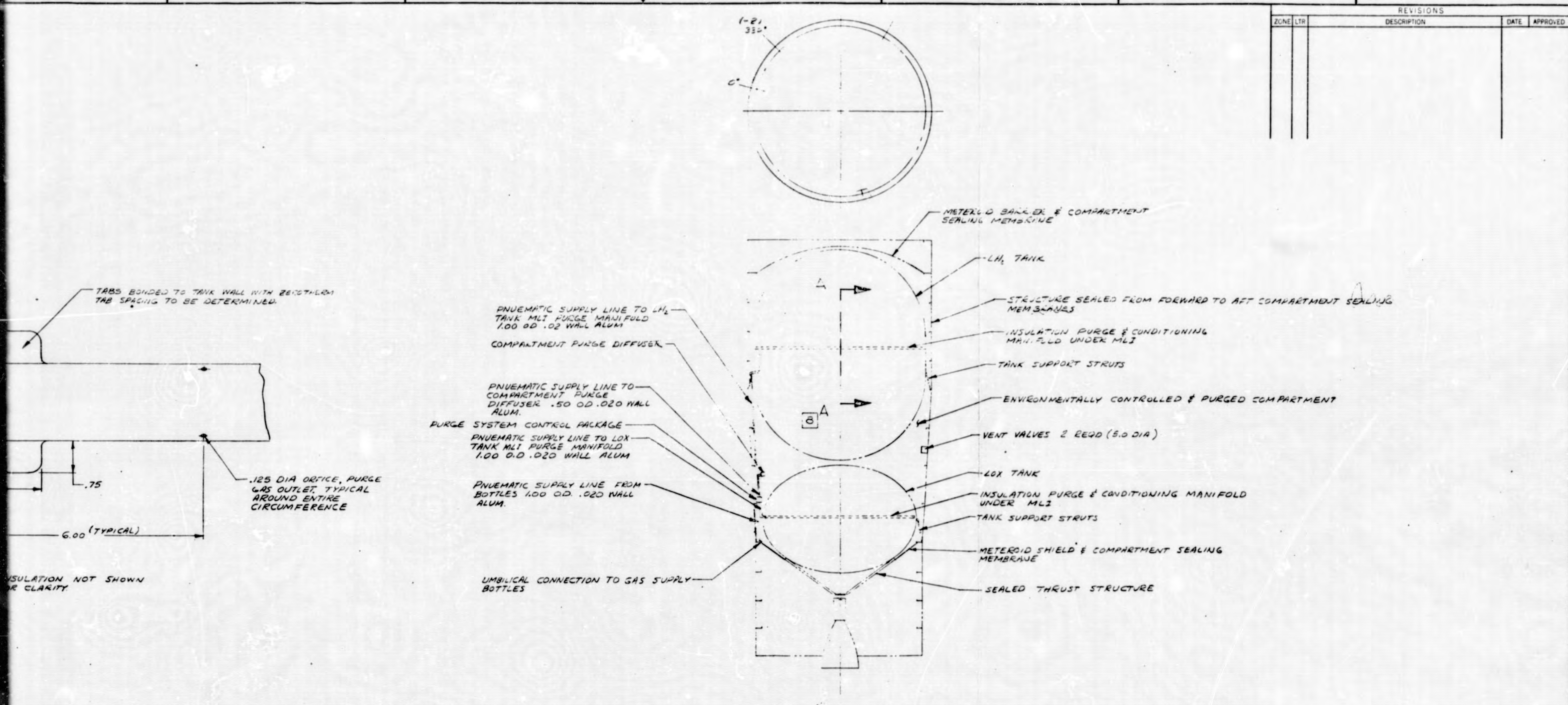
MANIFOLD TYP FOR LH_2 & LOX TANK

INSULATION NOT SHOWN FOR CLARITY



ORIENTATION VIEW
SCALE: NONE

F-1



ORIENTATION VIEW
SCALE: NONE

Figure 6.4-8
Concept Dwg-Conditioning & Purge System, Multi Layer Insul Tug (MCR 7007)

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CONCEPT DWG-CONDITIONING & PURGE SYSTEM, MULTI LAYER INSUL, TUG (MCR 7007)

SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	V7-923616

SCALE NOTED SHEET 1 OF 1

F-2

An evaluation of the available plastic film materials revealed that three different plastic film materials are available which have higher recommended service temperature than the polyester film. These films are compared with polyester film in the following chart:

Material	Max. Recommended Service Temperature	Minimum Available Thickness	Reference
FEP Teflon	440 F	0.0005 inch	6-7
TFE Teflon	500 F	0.00015 inch	DuPont Corp.
Polyimide (Kapton)	750 F	0.00025 inch	6-7
Polyester (Mylar)	300 F	0.00015 inch	6-7

The polyimide film appears to be the most suitable replacement for polyester film where temperature above 140 F make polyester film unacceptable. The polyimide film has the highest service temperature of all the known plastic films. Although the TFE Teflon film has a relatively high recommended service temperature and is available in as thin a gage as the polyester film, the strength of TFE Teflon film is only approximately one tenth that of either polyester or polyimide. For this reason the TFE Teflon film is considered too weak, in the thinner gages, to withstand normal service loads without rupturing. Also both the TFE and FEP Teflon films are more difficult to coat with a tightly adherent reflective metal surface than is the polyimide film.

Another characteristic which favors the selection of the polyimide is its low creep rate under stress at temperatures above 140 F. Although both the TFE Teflon and polyimide materials are considered to be thermosetting, the polyimide exhibits considerably less creep under stress at elevated temperatures. Both FEP Teflon and polyester materials are thermoplastic and flow readily under stress when heated. It is the high creep rate of the polyester film which causes it to lose the embossment pattern when heated above 140 F under compression and not any degradation of the polyester film material itself. Therefore, it is reasonable to expect the thermosetting polyimide film will retain its embossment pattern much better than other available films when heated above 140 F under compression. The low-creep characteristics of the polyimide film make it more difficult to emboss than films made from thermoplastics. However, polyimide can be formed and embossed at temperatures near its upper service temperature as demonstrated by films embossed with a "cut-bead" pattern by the Coburn Coating Co., Mt. Vernon, New York, and films formed into hemispheric shapes by the Schjeldahl Co., Northfield, Minnesota.

Polyimide film can be coated with vapor deposited aluminum to produce a highly reflective surface using the same procedure as has been used in the past to aluminize polyester (Mylar) films. As with all thin coatings of pure aluminum, vapor deposited aluminum coatings on polyimide film will corrode and

lose reflectance if they contact water such as may be formed by condensation from humid air. To prevent this possibility the multiple-layer-insulation system must be purged with dry gas and sealed within a dry-gas atmosphere during ground storage. This corrosion problem can be eliminated by replacing the aluminum coating with a reflective coating of a corrosion-resistant metal such as gold. However, gold coatings cost considerably more than aluminum coatings. Silver is not a satisfactory coating material because it tarnishes and loses its reflectivity. Various top coatings which will protect the silver against tarnishing or will protect an aluminum coating against corrosion are unsatisfactory because all of the known top coating materials degrade the reflective characteristics of the metal at cryogenic temperatures (10 to 40 microns wavelength). Also, the added weight of even a very thin protective coating would result in a substantial increase in the density of the reflective film material.

The selection of single aluminized polyimide (Kapton) film as an acceptable material for use in fabricating in Tables 6.6-1 and 6.6-2. TABLE 6.6-1 shows that the use of single aluminized polyimide film adds a weight penalty of only two pounds over the use of a single aluminized polyester film while meeting the increased temperature requirements imposed by the Tug vehicle.

The selection of polyimide film as a replacement for polyester film for cryogenic insulation on the Space Tug has made it necessary to establish a new method for bonding the film to itself during manufacturing. Polyester can be heat-bonded to itself. This characteristic provides a convenient method for quickly bonding together separate pieces of aluminized polyester film during fabrication of the film into multiple layer insulation for cryogenic fluid tanks. However, polyimide film cannot be heat-bonded to itself. Because there is a large demand for a heat-bondable polyimide film, the DuPont Company manufactures a form of polyimide film, designated Kapton Type F, which is coated on one surface with FEP Teflon resin to produce a heat-bondable surface. However, this type of material is unsuitable for use as multiple layer insulation because the weight of the Teflon resin coating causes an unacceptable increase in the insulation density.

Several different types of adhesives are recommended by DuPont for bonding polyimide film. None of these adhesives are considered acceptable for use in fabricating cryogenic insulation systems because they all require either a heat cure for at least twenty minutes or a much longer room-temperature cure. It, therefore, appears that the most acceptable method for bonding polyimide film to itself is to apply an FEP Teflon dispersion coating locally to cut sections of film at the positions where the material is to be bonded during fabrication and installation of the insulation. The material can then be heat bonded easily and quickly using a conventional hand-operated heat sealer. The FEP Teflon dispersion coating can be applied to local areas of polyimide film either by spraying or brushing. Teflon coatings applied by this procedure do not adhere to film as well as commercially applied heat-bondable coatings which usually are heat-cured onto the film. However, these dispersion coatings do adhere well enough to remain

Table 6.6-1 Insulation System Selection

MATERIAL	PARAMETERS: EMISSIVITY METAL	PARAMETERS: EMISSIVITY FILM	PARAMETERS: WEIGHT 25-30 GA	TEMP. LIMIT	TUG POINT DESIGN: CRYOGEN BOIL-OFF ASCENT	PHASING	ORBIT	TOTAL	SYSTEM WEIGHT	TOTAL WEIGHT
Single Aluminized Polyester (MYLAR)	0.035 to 0.045	0.30	0.00185 $LB/FT^2/SH$	+ 140F Under Compressive Ld	98 LB (1 HR)	29 LB (13 HR)	158 LB (126 HR)	285 LB	213 LB	498 LB
Single Aluminized Polyimide (Kapton)	0.035 to 0.045	0.40	0.00216	Over +300F	98 LB	29 LB	158 LB	285 LB	215 LB	500 LB
Single Goldized Polyimide (Kapton)	0.020 to 0.035	0.40	0.00285	See SAK	98 LB	27 LB	145 LB	270 LB	250 LB	520 LB

Table 6.6-2 Insulation System Selection

MATERIAL	CURRENT AVAILABILITY	CORROSION POTENTIAL: SUSCEPTIBILITY	CORROSION POTENTIAL: PREVENTATIVE ACTION	DEVELOPMENT TO 1976 STATE-OF-THE-ART
SAM	AVAILABLE FROM 2 QUAL. SOURCES	TEST SHOWS CORR. AFTER 9 MONTHS IN EXTREME ENVIRONMENT AT KSC	MLI IS PROTECTED BY TUG STRUCTURE & BY ORBITER. KEEP IN DRY PURGE & PROVIDE PRECONDITIONING FOR ACCIDENTAL INTRUSION OF MOISTURE	OPTIMIZE EMBOSSMENT PROCESS
SAK	LIMITED EXPERIENCE	SEE ABOVE	SEE ABOVE	EMBOSSING PROCESS
SGK	LIMITED FLAT SHEETS - NO EMBOSSED MATERIAL PRODUCED	1. NO CORR. FROM MOISTURE. 2. SUSPECT ELECTROLYTIC CORR.	1. DESIGN TO PRECLUDE CONTACT GOLD TO ALUMINUM. 2. PURGE TO PREVENT INTRUSION OF MOISTURE.	1. METALLIZING PROCESS TO ENSURE LOW ϵ AFTER EMBOSSMENT.

intact during normal handling of the film in fabricating the insulation system, and following the heat-bonding operation the FEP Teflon can be expected to exhibit very good adhesion to both film surfaces with which it is in contact.

6.6.2 Tension Membrane Fabric Selection

The tension membrane fabric is used to prevent the very weak multiple layer insulation films from ballooning and tearing during purging of the insulation or evacuating the air from the insulation during space flights. The textile fiber used for the tension membrane must be very stable dimensionally at the expected service temperature of 200 F for the Space Tug. Also, fabric woven from this fiber must have relatively low elongation at service tensile loads and should be highly resistant to flexing and abrasion. Although glass fiber cloth will meet the low elongation and dimensional stability requirements for the tension membrane, it does not have adequate flex and abrasion resistance to be considered completely reliable for this application.

Other textile fibers which were considered for this application were nylon, high-temperature nylon (Nomex), fluorcarbon, polyester (Dacron), various acrylics, various vinyl derivatives, polyolefin, and cellulose. A comparative evaluation was made of the physical characteristics of each of these fiber materials, and the Nomex fiber was found to have the most acceptable properties for use as a tension membrane fabric. The flex and abrasion resistances of Nomex compare favorably with both nylon and polyester (Dacron). Nomex has a lower elongation under load than nylon and about the same elongation as low-elongation polyester. However, the low elongation of polyester fabric is obtained by heat setting which causes the fabric to shrink considerably when heated above approximately 140 F. The shrinkage of Nomex, on the other hand, is only about 0.1 percent at 200 F. If necessary, Nomex can be treated with formic acid at room temperature which causes an immediate shrinkage of about 7 percent but produces complete dimensional stability of the fiber to further shrinkage at temperatures below 650 F. Nomex is commercially available in a number of highly porous weave patterns.

6.6.3 Inner Support Screen

The multiple layer insulation must be supported a slight distance away from the cryogenic tank in order to provide a path for the dry purge gas to pass under and up through the insulation during purging operations. Aluminum screening was selected for use as a support material for the insulation for several reasons. It can be obtained in weaves which are both highly porous to airflow and sufficiently rigid to afford adequate support for the insulation system. Aluminum screen has the same thermal expansion characteristics as the aluminum tank, and it does not cause bimetallic corrosion in locations where it contacts the tank material. Also, aluminum screen is lightweight and is easily stretch formable.

6.6.4 Cryogenic-Insulation Support-Post Assembly

The multiple layer insulation is prevented from shifting on the cryogenic tank by fixed support posts. These posts pass through holes in the insulation blanket but are not otherwise fastened to the insulation. The posts will be molded from a glass-fiber-reinforced epoxy resin. This material was selected for use in making the support pins on the basis of past service experience on the Apollo space program. Glass-fiber-reinforced epoxy components were used for low-temperature applications on this program and it was demonstrated that they maintained good structural properties at liquid hydrogen temperatures. The posts are to be attached to the liquid hydrogen tank walls using materials and procedures which, also, were used on the Apollo space program where they were demonstrated to be structurally acceptable at liquid hydrogen temperatures. This attachment system consists of bonding "hard spot" onto the tank surface with a low-temperature modified epoxy adhesive. The hard spots are cut from heat-resistant glass-fiber-reinforced phenolic plastic honeycomb core filled with polyurethane foam. Metal inserts are potted into the hard spots with low-temperature polyurethane resin mixed with nylon fibers. The support posts are threaded on one end and are attached to the hard spots by being screwed into the metal inserts.

6.6.5 Exterior Thermal Control Paint

In order to prevent an excessive temperature rise of the exterior surface of the Tug vehicle in space, a surface coating is required which will exhibit a solar-energy absorptance (α_s) which is at least four times as great as its service-temperature emittance (ϵ). This requirement is usually stated as α_s/ϵ must equal 0.25 or less. The coating material selected for this application is a clear room-temperature-vulcanizing methyl silicone elastomer pigmented with zinc oxide powder. This coating material was selected both because it has an α_s/ϵ of approximately 0.21 and because it has good resistance to degradation resulting from exposure to either weathering or ultraviolet radiation. Also, this coating was used on both the Mariner 5 and Saturn IV vehicles.

6.6.6 Interior Surface Sealant Coating

The interior wall surface of the Tug vehicle is made of epoxy-resin-impregnated glass-fiber cloth. These wall surfaces are slightly porous, and they will leak purge gas in the areas where the walls are used as an outer shroud cover for thermal insulation of cryogenic fluids. In order to minimize excessive leakage of the purge gas it will be necessary to coat the interior wall surface with a sealant coating. Two different coating materials have been found to be suitable for this application. These are an epoxy resin sealant coating and a room-temperature-vulcanizing methyl-silicone elastomeric coating. Neither of these coating materials exhibits characteristics which make it clearly superior or inferior to the other. Final selection of the optimum sealant coating for use on the Tug will be based on the results of future more definitive tests and evaluations.

6.7 INSULATION SYSTEM STRUCTURAL ANALYSIS

6.7.1 Structural Description:

The multi-layer insulation (MLI) system which is used on the Space Tug LOX and LH_2 Tanks utilizes embossed, singly aluminized Kapton for reflective shields. Each shield is approximately 22 inches in width and extends from the top to the bottom of each tank. The shields are arranged in modular form, five shields to a module, each module overlapping the module below it. Six modules are used, giving a total thickness of MLI of 1/2 inch. The MLI system is attached to the tank by circumferential rows of fiberglass posts, each supporting the load of the insulation below it. Slack is provided in the material between each row of posts so that no load is transmitted through the material beyond the posts that support it. A separate outer shield functions as a tension membrane to support the pressure loads of the lower shields.

The MLI system is supported approximately one inch off the tank wall by bonded hard spots and aluminum wire mesh. The posts are attached to the hard spots. The void between the tank wall and the lattice forms a plenum into which dry heated purge gas is ducted. The purge gas then flows radially outward through perforations in the MLI to remove moisture or other contaminators. This preconditioning is carried out in ground operations prior to loading of cryogens.

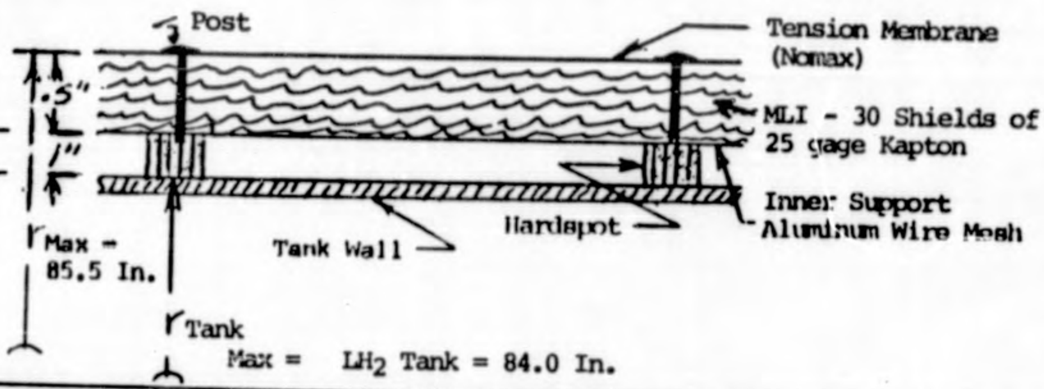
The Tug body shell structure functions as an external purge system, and protects the fragile shield material from mechanical damage and to prevent the intrusion of moisture or other contaminates. Each end of the Tug is closed off with a membrane to complete the purge system. A series of valves are installed to close off the purge system during storage periods and to permit venting during ascent.

6.7.2 Structural Analysis Procedure:

An analysis was performed of the structural elements of the internal MLI support system, the MLI, and the tension membrane to demonstrate the functional integrity of the system under operational and environmental loading conditions. Straightforward, established, analytical procedures were used together with simplifying assumptions and conservative analytical techniques. The properties of materials used in the fabrication process were taken from References 1, 2, and recent research and development testing in support of MLI Systems. These data may be readily substantiated. Load conditions were developed from the matrix shown as Table 6.7-1 Results of the analysis under these conditions indicate positive margins of safety of all structural elements.

Analysis:

The analysis presented is for the MLI and its support system. The prime consideration for the structural design is the differential pressure due to the purge and the ability of the structural components to support the weight



OPERATION		LH_2 TANK			LO_2 TANK			MAXIMUM ACCELERATION G's
		TANK		INNER SUP	TANK		INNER SUP	
		TEMP.	PRESS.	TEMP.	TEMP.	PRESS.	TEMP.	
GROUND	TANK PRESSURE TEST	AMB. -423 F	24 PSIA 24 PSIA	AMB. -423 F	AMB. -297 F	24 PSIA 24 PSIA	AMB. -297 F	± 1.0
**	MLI PRE-CONDITION	AMB.	AMB.	140 F	AMB	AMB	140 F	± 1.0
	CRYO LOADING	-423 F -423 F	AMB. AMB.	AMB. -423 F	-297 F -297 F	AMB. AMB.	AMB. -297 F	± 1.0 ± 1.0
	PRE-LAUNCH & STATIC TEST	-423 F	21 PSIA	-423 F	-297 F	21 PSIA	-297 F	± 1.0
LAUNCH **	LAUNCH	-423 F	15 PSIA	-423 F	-297 F	15 PSIA	-297 F	+ 3.3
ORBIT	OPERATION	-423 F	24 PSIA	-423 F	-297 F	24 PSIA	-297 F	± 1.0
	OPERATION	-423 F	20 PSIA	-423 F	-297 F	20 PSIA	-297 F	± 1.0
	TANK SAFING	-423 F	2 PSIA	-423 F	-297 F	2 PSIA	-297 F	± 1.0
	TANK SAFING	-237 F	15 PSIA	-423 F	-160 F	15 PSIA	-297 F	± 1.0
RE-ENTRY		-237 F	15 PSIA	-237 F*	-160 F	15 PSIA	-160 F*	+ 4.0

* Temperature of outer surface of MLI for this condition is +200 F.

** Maximum pressure drop (ΔP) across the MLI is during these conditions and is +0.0108 psi. ΔP for all of the other conditions is insignificant.

Table 6.7-1 Load Condition Derivation Matrix (MLI)

of the elements during acceleration loadings (G), as indicated in the following pages. Maximum support post spacing and structural dimensions occur on the LH_2 tank. FIGURE 6.7-1 is a schematic of the LH_2 tank insulation system.

When the MLI is purged, a delta pressure of 1.08×10^{-2} psi is generated across the MLI. This pressure is reacted by the outer membrane and transferred to the support posts. Additional loads are applied to the posts, and hard spots by the weight of the MLI multiplied by the load factors N_x, N_y, and N_z. A factor of safety of 1.4 is used for this analysis.

MLI

The MLI is checked for tear out of the unreinforced holes provided for the support posts under "G" loading, and the fusion joint under purge and "G" loading. The ultimate running load on one sheet of Kapton is 0.0034 lb/in which can be compared to a minimum test allowable of 0.0935 lb/in. The purge condition is critical for the fusion joint. Maximum ultimate load is .94 lbs and the minimum test allowable at elevated temperature is 4.0 lbs.

Support Post

The post detail is shown on Drawing V7-923612 zone 10. Critical loading is developed when the tension membrane forces are transferred to the post. Because a "perfect fit" of the tension membrane cannot be guaranteed, it must be assumed that 50 percent of the forces are unbalanced, thus producing bending in the post. The ultimate bending moment at the post base is 7.05 inch lbs and the minimum test allowable is 9.0 inch lbs.

Hardspot

All load conditions generate very small shear and tension stress on the hardspot bonded attachment to the tank compared to actual strength.

Tension Membrane

Maximum loading is the (P)(r) loading during purge. Maximum ultimate loading is 1.29 lb/in. A detail allowable load at 140°F for the Nomex membrane is not available. However, when the room temperature allowable of 55.3 lb/in is adjusted by the single fiber strength loss at 140°F, the change is less than 10 percent. Therefore, no problems are associated with the membrane.

Support Screen

Maximum loading is developed by "G" forces at re-entry on the weight of MLI supported by the screen. The screen must react this load in membrane bending as a flat rectangular plate supported at the four corners. The maximum ultimate stress developed is 217 PSI, which can be compared to the yield of annealed 6061 aluminum of 5,000 PSI.

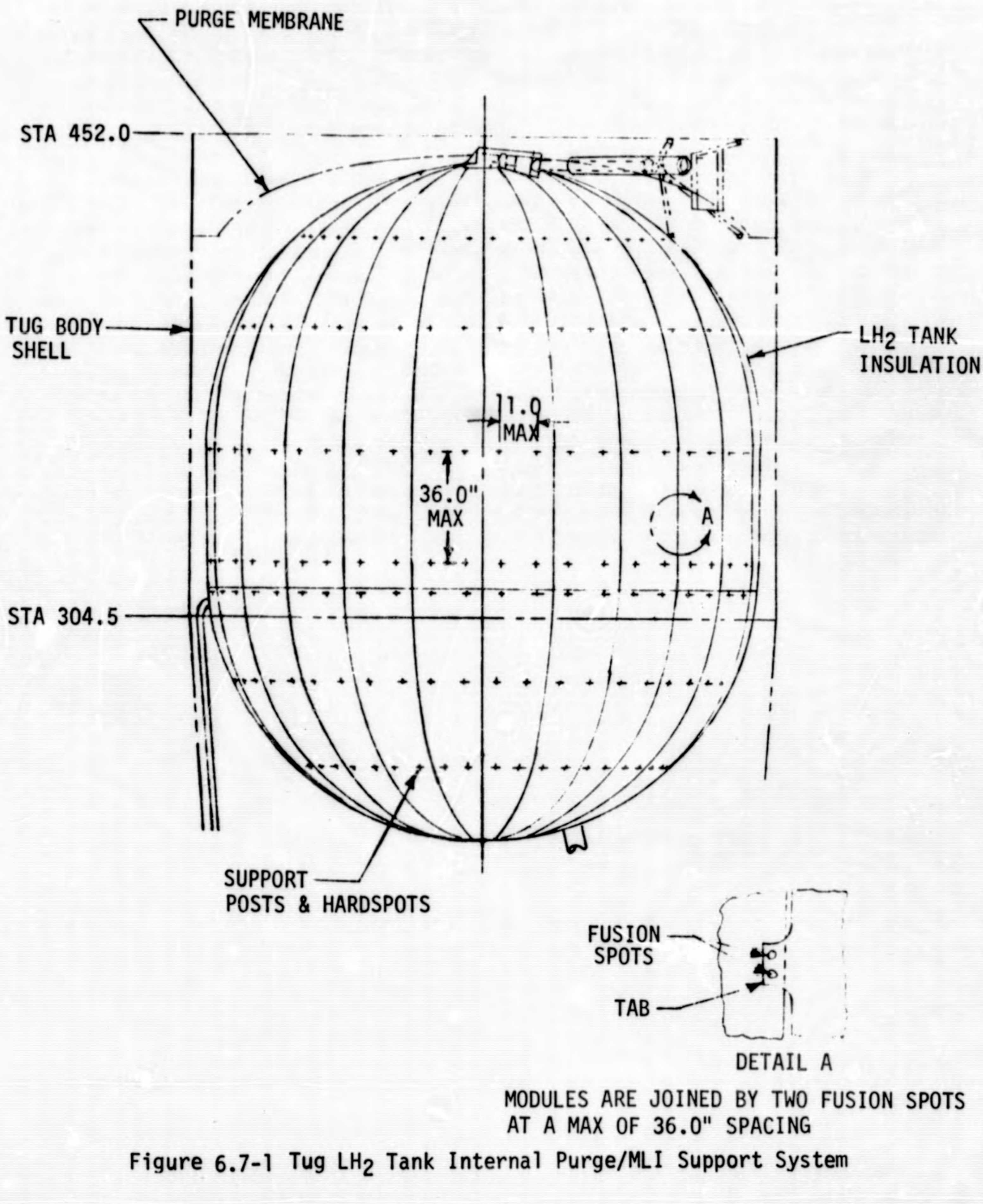


Figure 6.7-1 Tug LH_2 Tank Internal Purge/MLI Support System

6.8 TEST PROGRAM

Historically, cryogenic insulation design has followed an empirical approach in which potential concepts were established and then scale size specimens were fabricated and tested in the laboratory. Depending upon the results of such tests, the configurations were changed and additional tests conducted until an acceptable design was chosen. This empirical technique has in too many cases resulted in late identification of design problems, frequently not until after the vehicle is in the field where changes are costly or perhaps compromising to interfacing systems.

NR utilizes an analytic approach for cryogenic insulation design. This approach requires that authoritative data on materials properties be obtained for the anticipated environmental ranges at the start of preliminary design. Design details are then formulated and analyzed from this data utilizing classical thermal and structural techniques. The most appropriate details or combination of details are then fabricated and tested to verify the analysis. This approach provides sound engineering design, permits analytical evaluation of potential design details, eliminates the need for costly tests of design elements that may later be rejected and lends confidence to success of later large-scale system performance, verification and qualification. It also permits formulation of alternate and backup designs and determination of effects of off-design or contingency conditions.

Good design practice requires supporting tests in six general categories as shown in the matrix of Figure 6.8-1. The tests, while not described in detail are included in the overall test plan presented in Section 4 of Volume IV.

TEST PROGRAM	TYPE OF TEST	NATURE OF TEST/SPECIMEN	REASON
MATERIAL PROPERTIES TESTS	MONOSTRAIN - TENSILE COMPRESSION BENDING THERMAL CONTRACTION/EXPANSION TEAR-OUT STRENGTH LAYER DENSITY OPTICAL PROPERTIES MATERIAL DRAPE CALORIMETER	LABORATORY TESTS ON SMALL SPECIMENS USING STANDARD LABORATORY/TYPE EQUIPMENT. TESTS WILL COVER TEMPERATURE RANGE.	DESIGN OF INSULATION SYSTEM DETAILS
DESIGN AND PROCESS DEVELOPMENT	CALORIMETER STRUCTURAL ATTACHMENTS EVACUATION THERMAL SHOCK VIBRATION ACOUSTIC	TESTS OF SPECIFIC DESIGN DETAILS UNDER APPLICABLE ENVIRONMENT. SPECIMEN SIZE DEPENDENT UPON SPECIFIC TEST OBJECTIVES. EVALUATE LIFE CYCLE OF SPECIFIC DESIGN DETAILS.	EXPERIMENTAL ASSURANCE THAT DESIGN DETAILS WILL PERFORM AS ANALYZED. PROVIDE INFORMATION FOR TRADE-OFF OF DESIGN DETAILS. PROVIDE PERFORMANCE PARAMETERS. PROVIDE INFORMATION FOR PREPARATION OF PROCESS SPECIFICATIONS. DEVELOP INSPECTION PROCEDURES AND EQUIPMENT. DEVELOP REPAIR CRITERIA.
DESIGN/COMPONENT VERIFICATION	THERMO-VACUUM CALORIMETER THERMAL SHOCK VIBRATION ACOUSTIC	TESTS OF INSULATION CONCEPTS INCLUDING REPRESENTATIVE INSTALLATION & AREAS OF DESIGN CONCERN. CALORIMETER WILL BE FULL SIZE OR SCALED TO REPRESENT FULL SIZE.	VERIFICATION THAT FINAL DESIGN WILL MEET THERMAL OBJECTIVES AND IS STRUCTURALLY ADEQUATE. PROVIDE PERFORMANCE INFORMATION. VERIFY MLI REPAIR TECHNIQUES & PROCEDURES.
SYSTEM QUALIFICATION	THERMO-VACUUM CALORIMETER	FULL SCALE TANK-MLI SYSTEM INSTALLED BY MFG USING PRODUCTION MATERIALS, TOOLS, PROCEDURES UNDER Q.C. COGNIZANCE. TESTS UNDER DIRECTION OF TEST OPERATIONS, CLINICAL DISSECTION UNDER DIRECTION OF ENG. TWO CONSECUTIVE SUCCESSFUL TESTS REQ. FINAL TEST MAY BE TANKS FROM FIRST FLIGHT VEHICLE.	QUALIFICATION OF DESIGN, PROCESS PROCEDURES AND SPECIFICATIONS, MANUFACTURING INSTALLATION PERSONNEL. OBTAIN PERFORMANCE DATA.
INSTALLATION VERIFICATION	THERMO-VACUUM CHAMBER	FLIGHT VEHICLE-TANKS INSULATED IN FLIGHT CONFIGURATION. TESTS MAY BE CONDUCTED IN MULTIPLE TEST SITES	VERIFICATION THAT SYSTEM INSTALLATION IS SATISFACTORY FOR FLIGHT.
FLIGHT EVALUATION	FLIGHT TEST	FLIGHT VEHICLE - INSTRUMENTATION INSTALLED TO MONITOR PERFORMANCE	EVALUATION OF SYSTEM PERFORMANCE, CHARACTERIZATION OF PARAMETERS, PREDICTION OF PERFORMANCE ON SUCCEEDING FLIGHT.

Figure 6.8-1 Insulation System Test Philosophy

REFERENCES

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6-3. Cryo-Storage Thermal Improvement, NR SD 71-263, (February 1971)

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6-5. NASA/MSFC TMX 53872 Terrestrial Environment (Climatic) Criteria Guidelines For Use In Space Vehicle Development, March 15, 1970 Revision

6-6. NASA/MSFC TMX-53957, Space Environment Criteria Guidelines For Use In Space Vehicle Development (1969 Revision). Second Edition, August 26, 1970.

6-7. Modern Plastics Encyclopedia, 1970/1971, McGraw-Hill, Inc., New York.

7.0 METEOROID PROTECTION SUBSYSTEM

7.1 REQUIREMENTS

For meteoroid protection analysis, it was assumed that the six-day tug design mission was as described in Table 7.1-1 (as obtained from Reference 7.3-1) and that each vehicle would be designed for a useful life of 20 missions.

The meteoroid environment model applied was that of Reference 7.3-2, in which the meteoroids are of the cometary type, where density = 0.5 gram per cubic centimeter. The flux of these particles is assumed to be omnidirectional in undisturbed space and correction factors are applied to account for the shielding and gravitational effects of the earth. Cometary particles occur sporadically and in streams. For simplicity, an average total flux combining these criteria was applied; or

$$N = 10^{-14.37} G_e \zeta / m^{1.213} \qquad (1)$$

where:

N = flux of meteoroids equal to or larger than m, meter^{-2} - sec^{-1}

G_e = earth gravitational factor

ζ = earth shielding factor

m = meteoroid mass 1 grams

Meteoroid impact velocity was taken as 20 km/sec

It was required that the structure be designed to provide 0.95 probability or greater that no failure of the LOX or LH_2 tanks would occur during the design life due to meteoroid impact. The approach taken was to allow penetration into the tanks by meteoroid debris up to 25 percent of the tank wall thickness. It was required, therefore, that the tank design be compatible from a fracture mechanics standpoint with this level of damage.

7.2 ANALYSIS

7.2.1 Approach

Applying the results of earlier parametric meteoroid protection analysis (Reference 7.3-3) to the tug configuration, it was found that the meteoroid protection requirements were minimal, and it was evident that

Table 7.1-1. Tug Timeline

Time		Duration of Events		
Day	(Sec) Hrs	Hrs	No.	Event
0.0	0.00	0.00		Lift-off
	0.12	0.12	2	Inject into 50 by 100, 28.5° orbit
	0.85	0.73	3	Coast to 100 nmi apogee
		0.00	4	Circularize Shuttle and payload
	2.85	2.00	5	Undock Tug and back off from Shuttle
	14.85	12 00	6	Phase in orbit for longitude (0-12 hours)
		0.00	7	Burn and inject into 100 by 19,300 nmi transfer orbit Change plane 2°
0.84	20.13	5.28	8	Coast to 19,300 nmi apogee
		0.00	9	Circularize and change plane 26.5°
	21.13	1.00	10	Deploy payload in orbit
3.88	93.13	72.00	11	Phase to retrieve other payload in orbit
	95.13	2.00	12	Rendezvous with payload
4.00	96.13	1.00	13	Dock Tug and payload
	96.63	0.50	14	Safe payload
4.53	108.63	12.00	15	Phase in orbit to return to Shuttle (0-12 hours)
4.53	108.63		16	Burn and inject into 19,300 by 270 nmi transfer orbit Change plane 26.5°
4.74	113.95	5.32	17	Coast to 270 nmi perigee
		0.00	18	Circularize Tug and payload change plane 2° to 28.5° inc orbit
5.66	135.95	22.00	19	Phase in orbit to return to Shuttle (0-22 hours)
		0.00	20	Burn and inject into 270 by 100 nmi transfer orbit
5.70	136.71	0.76	21	Coast to 100 nmi perigee
		0.00	22	Circularize tug and payload
5.78	138.71	2.00	23	Tug and Shuttle rendezvous
5.82	139.71	1.00	24	Dock tug and Shuttle
6.45	154.71	15.00	25	Phase to return to launch site (0-15 hours)
		0.00	26	Deboost to return to Earth
6.48	155.41	0.700	27	Land at launch site (42 min)

(1) A single bumper type shield, rather than a dual bumper shield, should be applied.

(2) By considering the materials required for load carrying and the thermal insulation, little or no additional material would be required for meteoroid protection.

The approach taken was to utilize the aft skirt, the intertank structure, the forward structure, and the purge barriers as the meteoroid bumper and to employ the multilayer insulation as a meteoroid shield. The insulation thickness and/or density would be increased as necessary to limit tank penetration to 25 percent of the wall thickness. Fracture mechanics would be applied to verify that this level of tank damage could be tolerated.

7.2.2 Materials

To minimize structural weight, it was desirable to consider use of graphite-epoxy or glass-epoxy material for the structure and purge barrier members. Very little hypervelocity particle impact data are available on either of these materials. Based on work of Swift and Hopkins (Reference 7.3-4), it appears that the performance of these materials as bumper would be comparable to aluminum, which has been tested exhaustively and found to be a good material.

7.2.3 Minimum Bumper Thickness and Spacing

The function of the bumper is to completely fragment and scatter the meteoroid on impact so that the insulation and tank wall are subjected to impact by debris fragments of much reduced size. The minimum requirements for bumper thickness and spacing were computed to establish a lower limit for the tug shell and purge barrier structure.

To account for changes in altitude and the corresponding changes in meteoroid flux, the mission was divided into four phases as tabulated, and the flux correction factors were computed. The probability of no impact of a meteoroid of mass m or larger is given by Reference 7.3-5.

Phase (k)	Altitude NM	Time (T_k)		Correction Factors	
		Hrs	Sec	ζ_k	G_{ek}
1	100	320.0	11.5×10^5	0.606	0.990
2	19,300	1,218.6	44.0×10^5	0.992	0.640
3	19,300	1,218.6	44.0×10^5	0.992	0.640
4	100	320.0	11.5×10^5	0.606	0.990

$$P_o = e^{-\lambda} (\lambda > 0.1) \tag{2}$$

$$= 1 - \lambda \ (\lambda \leq 0.1)$$

where

$$\lambda = N(m) \sum_{k=1}^{4} G_{ek} \zeta_k A_k T_k \tag{3}$$

using equation 1 in 3

$$\lambda = 10^{-14.37} \sum_{i=1}^{4} G_{ek} \zeta_k A_k T_k / m^{1.213} \tag{4}$$

Substituting in 2 and rearranging, the nominal design meteoroid is given by

$$m = \left[\frac{10^{-14.37} \sum_{k=1}^{4} G_{ek} \zeta_k A_k T_k}{1 - P_o} \right]^{1/1.213} \text{ grams} \tag{5}$$

Using a tug area (A_k) of 140.7 square meters, $P_o = 0.95$, and the values tabulated above in equation 5, the nominal design meteoroid was computed to be 4.21×10^{-4} grams mass and 0.117 cm diameter.

Applying the theory of Reference 7.3-6, the minimum bumper would be sized so that the mass of material removed from this bumper by the meteoroid impact is 0.8 times the meteoroid mass. To meet this criteria, the thickness values were established as 0.009 and 0.007 inches for graphite-epoxy and glass-epoxy, respectively. Where honeycomb sandwich is used, the face sheet thickness should be at least half these values. Applying the same theory, the spacing between bumper and tank should be at least 30 times the meteoroid diameter to achieve the maximum benefit of spacing; therefore, minimum spacing was established as 1.5 inches.

The results of the structure design effort was that the thickness and spacing required to meet structural load requirements exceeded these minimum values; therefore, the baseline values were used in subsequent meteoroid analysis.

7.2.4 Insulation Sizing

7.2.4.1 Tug Construction

To account for variations in the facesheet thickness, facesheet material, and tank wall thickness; the baseline tug was divided into assemblies and components, as shown in Figure 7.21. The corresponding wall section data and

FIRST NO. DENOTES ASSEMBLY
SECOND NO. DENOTES COMPONENT

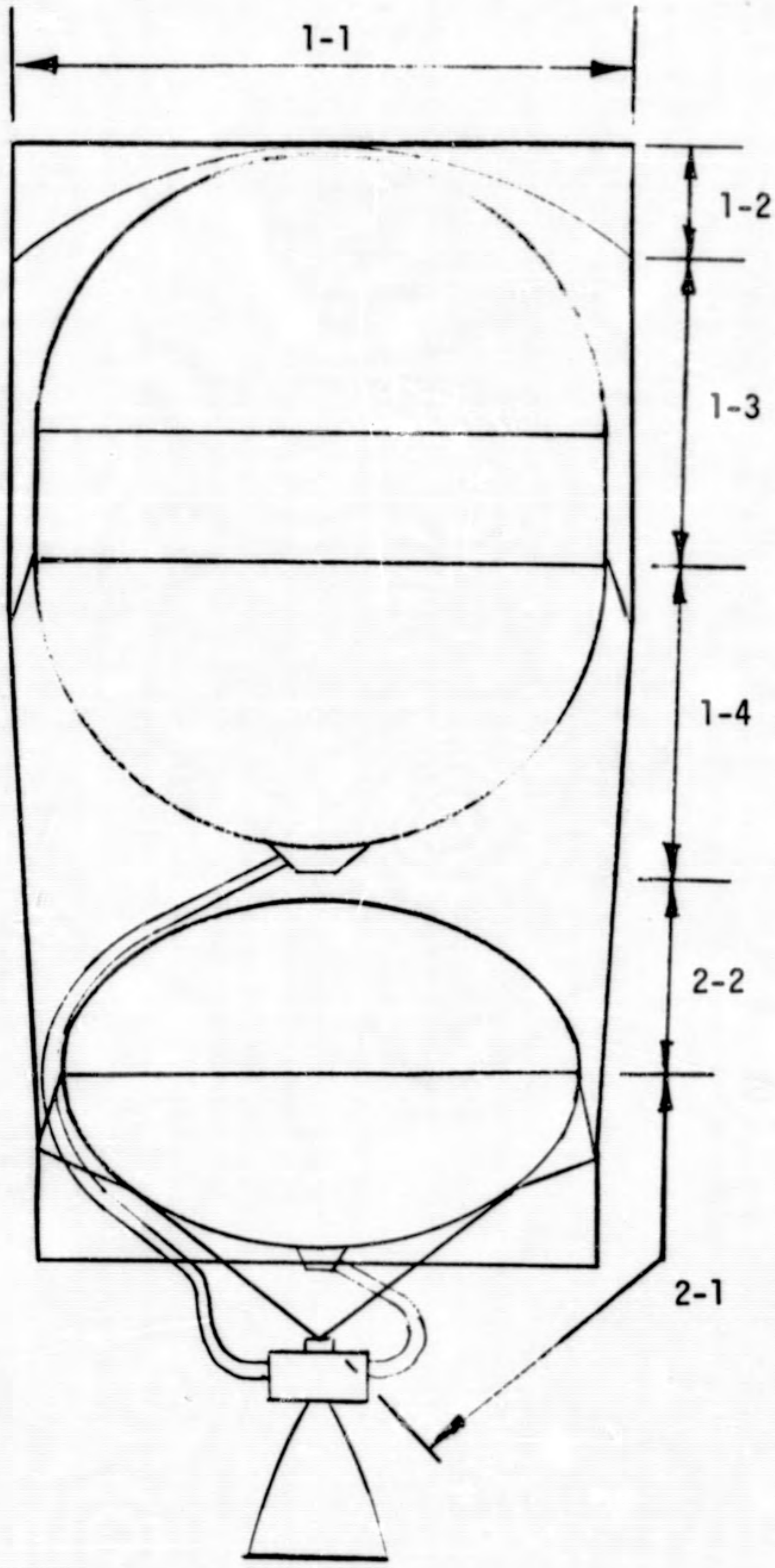

Figure 7.2-1 Meteoroid Analysis Surface Breakdown

vulnerable surface area are presented in Table 7.2-1. Insulation thickness and density were treated as variables and a range of values were selected.

7.2.4.2 Tank Penetration

Meteoroid masses ranging from 10^{-3} grams to 10^{-1} grams were selected; and for each assembly-component combination, the maximum depth of penetration into the tank wall was computed for various thicknesses of insulation. To make this computation, the penetration mechanics computer program of Reference 7.3-6 was employed. Tank penetration was found to be essentially independent of the honeycomb construction, within the limits of Table 7.2-1. Typical results are presented in Figure 7.2-2. Combining these data with the tank thickness values of Table 7.2-1 and allowing 25 percent wall penetration, the mass of the meteoroid to cause failure was established as a function of insulation thickness.

7.2.4.3 Probability of No Failure

The design mission was broken into four phases as described earlier to account for altitude changes with the result that the probability of no tug tank failure during n missions is given by

$$P_{NF} = \left(\prod_{k=1}^{4} P_{nfk}\right)^{n} \qquad (6)$$

The probability of no tank failure during phase k is

$$P_{nfk} = \prod_{i=1}^{2} P_{nfik} \qquad (7)$$

where i is the assembly number, the probability of no failure of assembly i during phase k is

$$P_{nfik} = \prod_{j=1}^{n} P_{nfijk} \qquad (8)$$

and, the probability of no failure of component ij during phase k is

$$P_{nfijk} = 1 - \lambda_{ijk} (\lambda \geq 0.1) \qquad (9)$$

when

$$\lambda_{ijk} = G_{ek} \zeta_k T_k A_{ijk} N_{ij} \qquad (10$$

N_{ij} = the flux of meteoroids which could fail component ij.

Table 7.2-1. Tug Wall Section Data[b][c]

Assembly No. (i)	Assembly	Component No. (j)	Component	Face Sheet[a]		Nominal Tank Thk, Inch	Vulnerable Area M^2
				Thk Inch	Material		
1	LO_2 Tank	1	Upper Dome, End	0.006	GLE	0.023	16.4
		2	Upper Dome, Forward Skirt	0.008	GRE	0.023	21.9
		3	Cylinder	0.008	GRE	0.023	34.1
2	LH_2 Tank	1	Lower Dome	0.008	GRE	0.033	34.4
		2	Upper Dome	0.008	GRE	0.043	19.2

(a) GRE = graphite-epoxy; GLE = glass-epoxy

(b) Spacing between shell and tank greater than 4.0 inches

(c) Tank material = 2014-T651

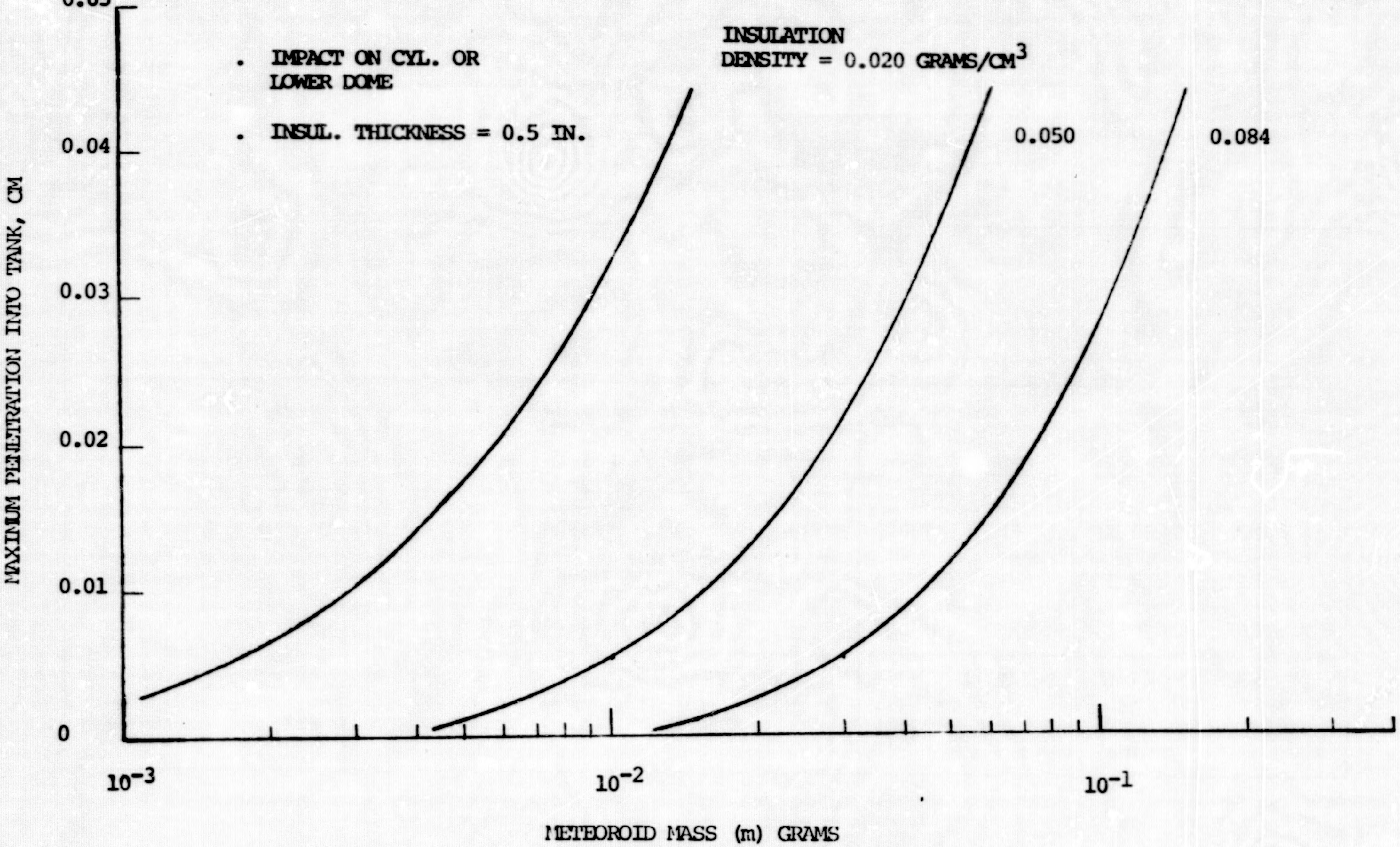

Figure 7.3-2 Predicted Meteoroid Damage to Tug LH_2 Tank

Using the failure masses obtained as described in 7.2.4.2 with equation 1, the failure flux values were obtained. These were used in equation 10 to obtain λ values for all combinations. The probability of no failure was then computed by solution of equations 6 through 9. Results are summarized in Figure 7.2-3.

7.2.4.4 Evaluation

The final insulation thickness and density for the baseline tug were established as 0.5 inches and 1.36 lb/ft^3. Converting units and interpolating in Figure 7.2-3, the probability of no failure of the baseline tug is greater than the required 0.95 value.

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1. Preliminary Design Office, NASA/MSFC, "Tug Point Design Study Plan," October 25, 1971.

2. Weidner, D.K., "Space Environment Criteria Guidelines for Use in Space Vehicle Development," (1969 Revision) 2nd Edition, August 26, 1970. NASA TM X-53957.

3. Anon. - "Pre-Phase A Study for Analysis of Reusable Space Tug," Volume 4, SD71-292-4, March 22, 1971.

4. Swift, H.F. and Hopkins, A.K., "Effects of Bumper Material Properties on Operation of Spacecraft Hypervelocity Particle Shields." Report No. AF-ML-TR-68-257, Sept. 1968.

5. Frost, V.C., "Meteoroid Damage Assessment," NASA SP-8040, May 1970.

6. Richardson, A.J. and Sanders, J.P., "Discrete Particle Penetration Mechanics Based on Debris Particles of Two Materials," SD-70-463. November 1971.

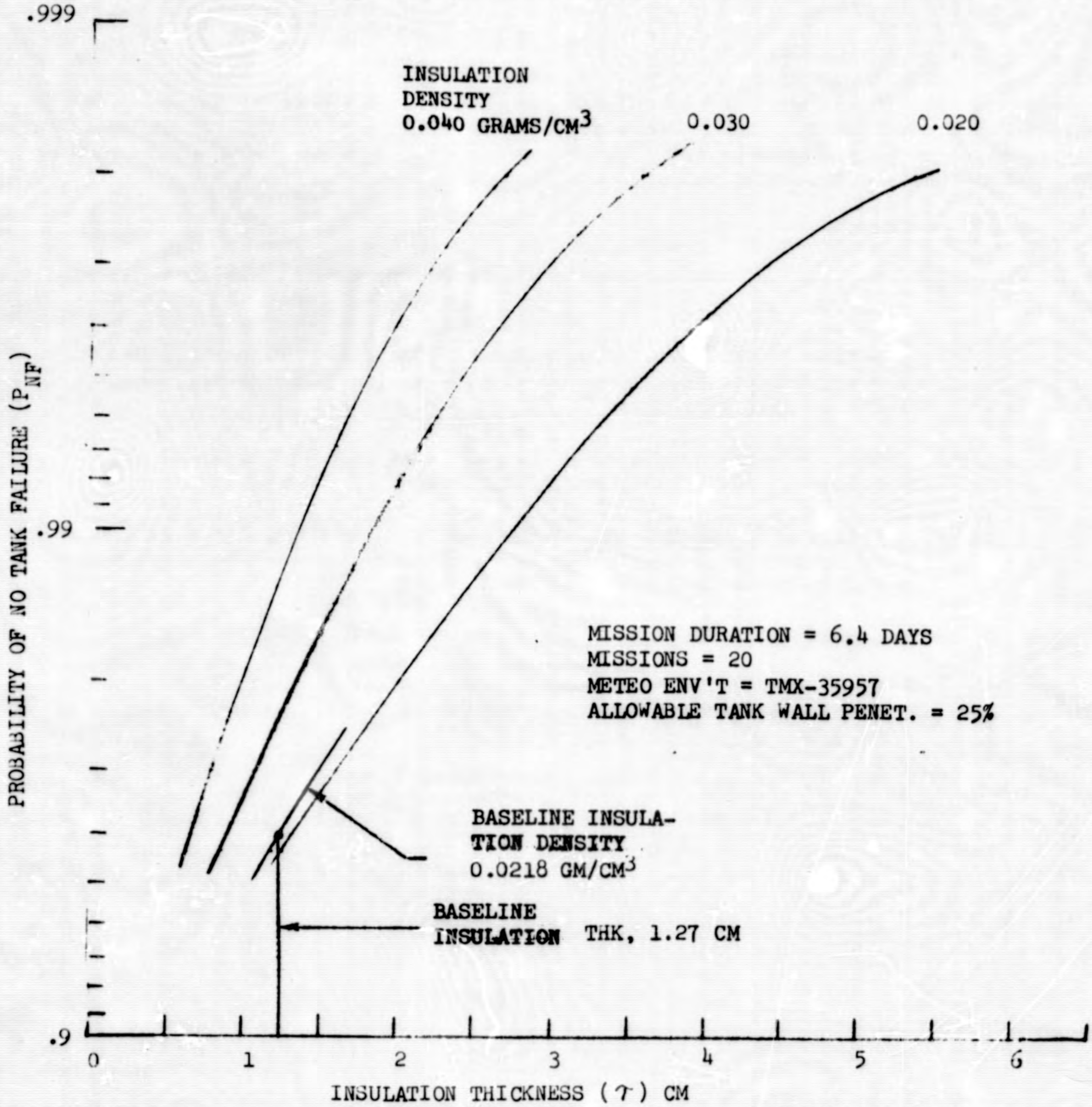


Figure 7.3-3 Effect of Thermal Insulation on Tank Meteoroid Protection

8.0 STRUCTURAL SUBSYSTEM

This section of the report defines the primary vehicle structure, Shuttle support structure, and the docking structure required to perform the Tug missions. Material selection and processing techniques considered for this vehicle are projected to a 1976 technology baseline.

Detailed calculations used in establishing sizes are not included in this report but are available for examination.

8.1 DESIGN CRITERIA AND MATERIAL PROPERTIES

8.1.1 Design Criteria

The following design criteria were used in defining the Space Tug structure.

a. All primary and secondary structural components where critical load occurs while the Space Tug is attached to the Space Shuttle will be designed with minimum allowable safety factors of 1.4 ultimate and 1.1 yield. For a structural component whose critical load condition occurs during Tug operation, or other time where failure of the component will have no effect on the Space Shuttle system, the component may be designed with minimum allowable safety factors of 1.25 ultimate and 1.05 yield.

b. The Space Tug propellant tankage will be designed employing fracture mechanics procedures specified in NAS SP-8040, "Fracture Control of Metallic Pressure Vessels," as a guide and based on the service life shown in Table 8.1-1 and Figures 8.1-1 and 8.1-2.

c. A safe life design philosophy will be followed in the design of all Space Tug structural components. All structural components will be designed for positive margins of safety for 20 mission cycles.

d. Structural material selection and properties will be based upon projected 1976 materials technology.

e. Required Space Tug handling and transportation fixtures will be designed such that no structural load conditions exceeding the flight environment are encountered.

f. Structural component designs will be reviewed and evaluated for applicability to refurbishment and periodic servicing to allow for reductions of safe life design requirements.

TABLE 8.1-1

Space Tug Pressure Service Life Requirements

Operation	No. of Cycles	Pressure psia LH_2	Pressure psia LOX	Temperature °F LH_2	Temperature °F LOX
Proof Test	1	26.2*	30.8*	R.T.	R.T.
Cryo Proof Test	1	33.6*	35.0*	-423	-320°F
Static Firing					
Pretest Checkout	1	24	24	R.T.	R.T.
Test	3	24	24	-423	-297
Post Test Checkout	1	24	24	R.T.	R.T.
Flight					
Preflight Checkout	20	24	24	R.T.	R.T.
Flight	20 missions	Figure 8.1-2	Figure 8.1-1	Figure 8.1-2	Figure 8.1-1
Post Test Checkout	19	24	24	R.T.	R.T.

*Proof Pressures are PSIG

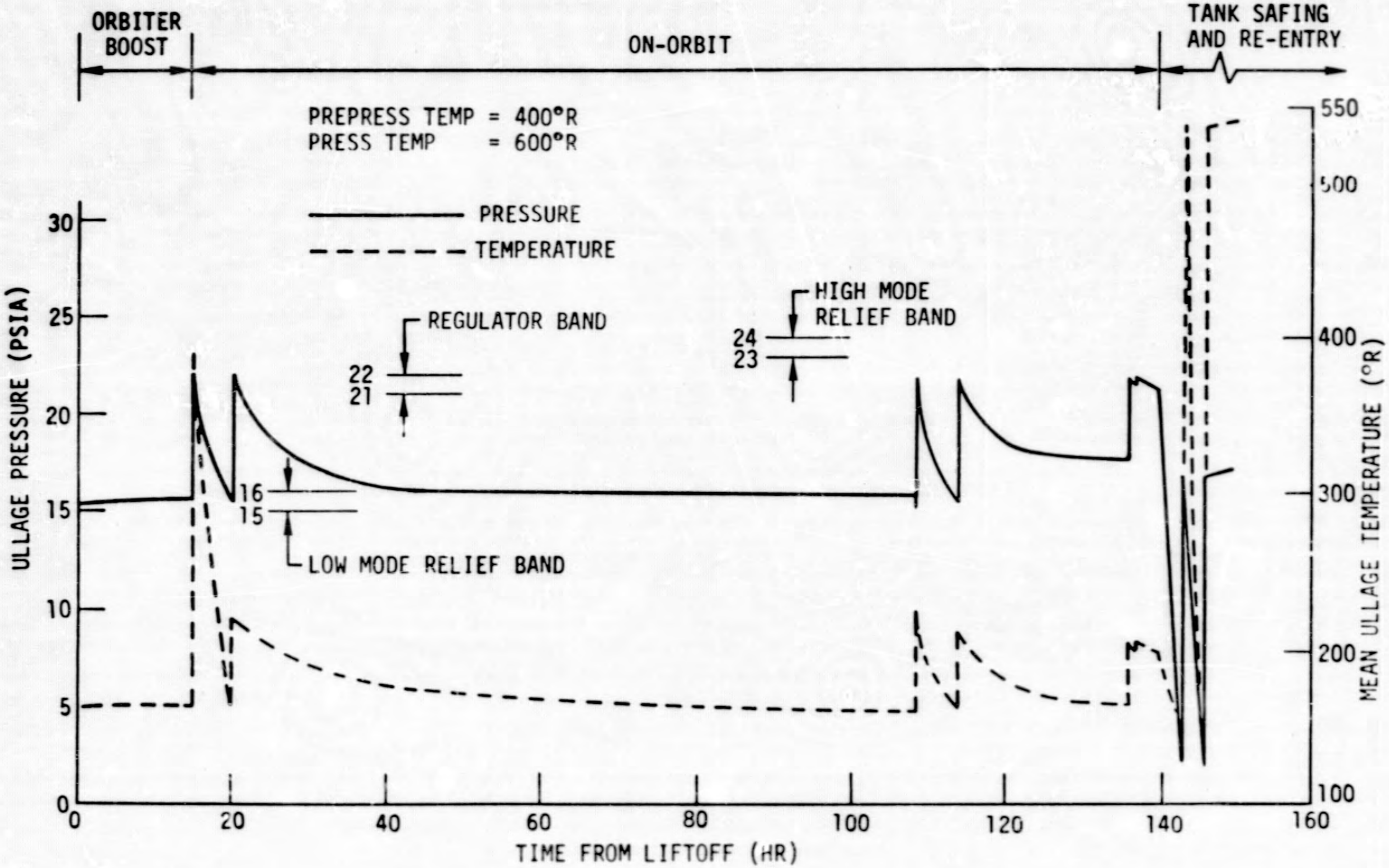


Figure 8.1-1 Main LOX Tank Temperature/Pressure History

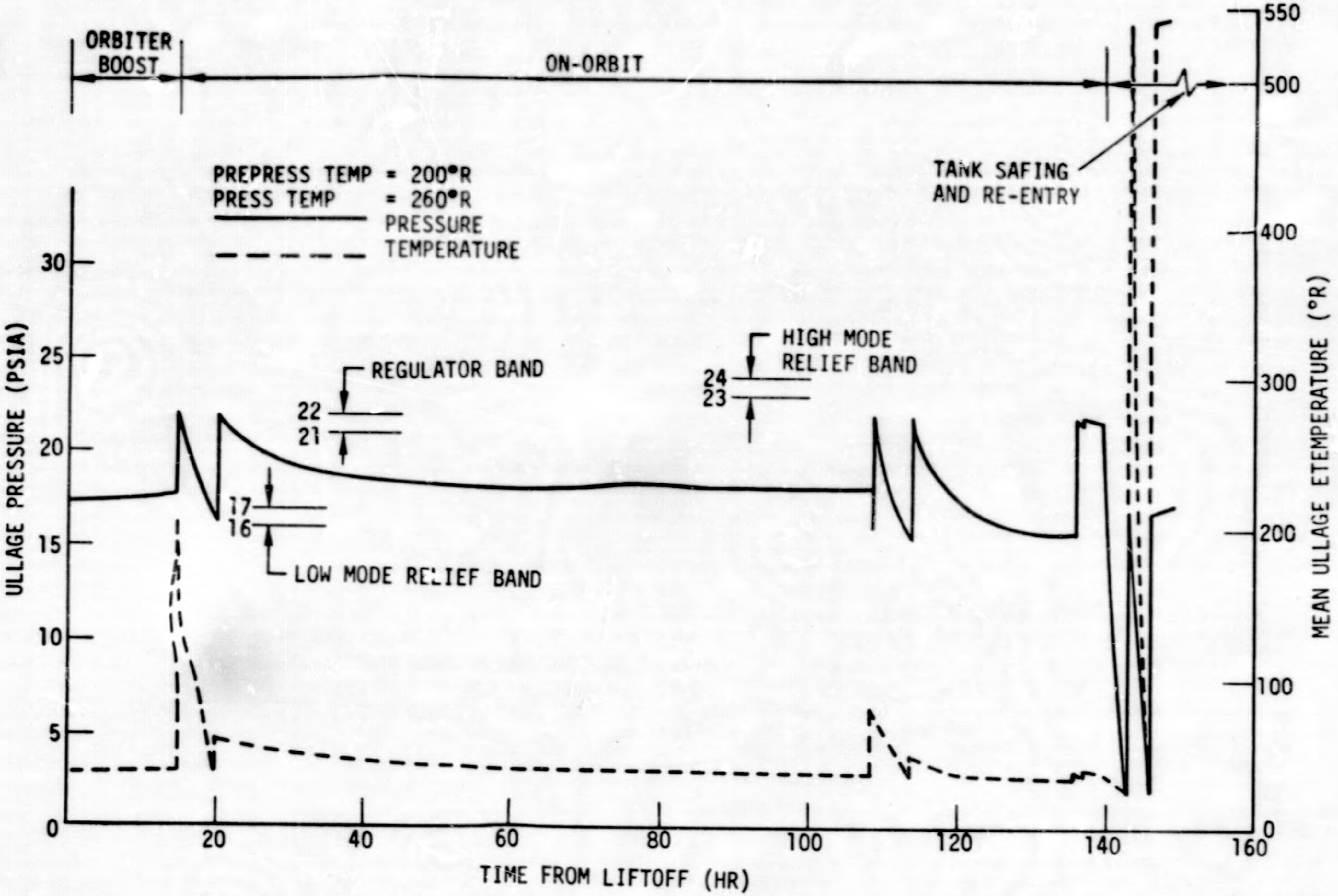


Figure 8.1-2 Main LH_2 Tank Temperature/Pressure History Preliminary

g. The baseline Tug will be designed to separate from the Shuttle at 100 n. mi. 28.5° with a 3k - 15' x 25' payload attached; ascent to geosynchronous orbit; deploy the up payload; retrieve a 3k - 15' x 25' payload within 6000 n. mi. (geosynchronous orbit) of the deployed payload; return to the Shuttle with the retrieved payload; redock with the Shuttle and return to Earth. The Tug will not provide any payload pointing or maneuvering services while on orbit except those required to transport the payload to or from its desired orbital station. The Tug will also be capable of separating from the Shuttle at 100 n. mi 28.5° with a 8.06k 15' x 25' payload attached; ascent to geosynchronous orbit; deploy the payload; return to the Shuttle for return to Earth. It shall also be capable of separating from the Shuttle at 100 n. mi. 28.5° with no payload; ascent to geosynchronous orbit; rendezvous and dock with a 4.16k payload; return with that payload to the Shuttle for return to Earth.

h. Primary structural support of the payload while in the Space Shuttle will be from the Space Tug/payload structural interface located at the forward end of the Tug. For long payloads, secondary structural supports from the payload to the Space Shuttle cargo bay support points may be incorporated to reduce excessive structural deflections and loads which would otherwise result. The secondary structural supports may be designed with spring-damper characteristics to preclude the introduction of Shuttle flight loads into the payload and Space Tug.

i. The Tug docking system design for both the Shuttle and payload docking operation must meet the limitations defined in the following Table.

DOCKING ACCURACY SPECIFICATIONS

	Structural	G&C
Centerline miss distance	0 to 1.0 foot	0. to 0.75 foot
Miss angle	0 to 5.0 degrees	0 to 1.0 degrees
Longitudinal velocity	0.1 to 1.0 ft/sec.	0.1 to 1.0 ft/sec.
Lateral velocity	0 to 0.30 ft/sec.	0 to 0.3 ft/sec.
Angular velocity (combined minimum of pitch, yaw and roll motion)	0 to 0.50 degrees/sec.	0 to 0.5 degrees/sec.

j. The Tug thrust structure will be designed for 10,000 pounds thrust with a seven degree gimbal angle.

8.1.2 Material Properties

Material selections considered for the Tug vehicle included both metallic and non-metallic materials. Selection of alternate candidate materials was based on achieving a minimum weight structure within the 1976 technology baseline. Material alternates for the propellant tanks were limited to filament wound tanks and aluminum alloys. Filament wound tanks were discarded early in the program because of questionable weight savings when considering both attachment requirements and the need for an internal liner. Additionally, the filament wound tanks in the size range necessary for the Tug vehicle cannot be projected to the 1976 technology baseline. Previous missile experience, reliability, and strength-to-weight ratio resulted in the selection of the aluminum alloys for the propellant tankage. The aluminum alloys considered were 2219, 2014, 2021 and possible combinations of the 7000 series materials. The 7000 series alloys with satisfactory strength properties were considered inadequate for the Tug design because of poor weldability and fracture toughness characteristics, especially in the cryogenic temperature range. The 7000 series alloys which are weldable such as 7039 and 7007 were eliminated because of their lower strength properties. Therefore, the tankage materials were reduced to the 2000 series for evaluation. The new aluminum alloy 2021 was eliminated due to the lack of production history and inferior fracture toughness characteristics as compared to the 2219 and 2014 materials. Also, no real benefits could be gained from the strength properties of this alloy. Although 2219 was superior to 2014 material in welding and stress corrosion characteristics, 2014 was selected for the tankage material, based on the higher strength properties in both the parent metal and weldments (especially in the cryogenic temperature range) and comparable fracture toughness characteristics in the thin material range. The design employed for the Tug tankage precludes serious stress corrosion susceptability problems by the avoidance of heavy sections. Weldability of this material is satisfactory and demonstrated on both the Saturn S-II and S-IVB stages.

Other metallic materials used in the Tug design were limited to end fittings and Longerons. Titanium was selected in these regions because of its high strength and system compatibility. In order to meet the weight requirements of the Tug design, the forward skirt, aft skirt, intertank structure and Tug to orbiter adaptor were designed utilizing advanced composite materials. Although the cryogenic design allowables for these materials were based on limited data, the "state-of-the art" technology for advanced composite materials was considered adequate for the Tug design. The forward skirt, intertank, aft skirt, and Tug to orbiter adapter structure are of honeycomb construction encompassing aluminum core with Graphite Epoxy face sheets. Alternates considered for these shell structures included Boron composites, aluminum alloys, and magnesium. Boron composites were not used because of the thin gages required. Boron filaments are limited to 0.005 in. per layer while Graphite can be produced to a thickness of 0.002 in. per layer. The four layers required to provide an isotropic facing sheet would result in a thickness of 0.020 in. versus 0.008 in. for Graphite. Conventional aluminum construction would result in a large weight penalty and magnesium would result in serious

corrosion problems during its twenty mission life. Gussets, frame web stiffeners and ring frames were also designed of Graphite Epoxy laminates due to their intricate design. Fiber ductility is required for parts of this nature eliminating the use of Boron and glass epoxy laminates. S-glass reinforced epoxy material was selected for the tank support struts because of its low thermal conductivity and high strength.

The thrust structure design requires a material with low thermal conductivity and adequate stiffness to preclude excessive engine deflections. A combined Boron/Graphite epoxy composite was selected for the tubular truss members. The Boron filaments, with their low thermal conductivity, are oriented longitudinally along the strut. Boron composites cannot be used in the circumferential direction because of the small bend radii required. The Boron filaments are brittle normal to the filament direction and tend to break when formed in small radii. Graphite epoxy was selected for hoop direction layers. Thermal conductivity of the Graphite composite is low normal to the filament direction. The resultant matrix has low thermal conductivity and high modulus of elasticity providing a rigid structure.

The metallic and advanced composite material design allowables are noted in Tables 8.1-2 and 8.1-3, respectively. It should be noted that the values specified for the advanced composite materials are preliminary and are subject to change as more data becomes available.

The composite material properties shown are based on 1972 technology. Approximately a 20 percent increase in allowables is expected by 1976. This increase in properties will be due to major advancements in fabrication and inspection techniques and uniformity of materials used in composite design. The 0.002 inch ply thickness for the Graphite fiber base Prepregs are current state-of-the-art and require no additional development beyond basic material properties.

The material property tables include the selected materials for the Tug design and alternate materials considered throughout the study.

8.2 BODY LOADS AND ENVIRONMENT

8.2.1 Body Loads

Critical loading conditions have been selected from the Shuttle Payload load factor conditions of Table 8.2-1. The critical loading conditions and the resulting attachment loads between the Tug and the EOS are presented in Table 8.2-2. Boost, landing and entry are the loading conditions considered. Space operations loads are not presented since they are not considered critical from a structural design standpoint.

During boost, the Tug is in the Space Shuttle cargo bay (mounted backwards). The Tug fitting configuration is presented in Table 8.2-2. Figure 8.2-1 shows the weight data used for the three configurations considered: a fueled Tug plus an 8060 lb payload, an empty Tug plus a 4160 lb payload, and a fully fueled Tug with no payload. The Tug body loads sign convention is shown in Figure 8.2-2.

TABLE 8.1-2.
DESIGN ALLOWABLES - METALLIC MATERIALS
Aluminum Alloys

Material	2219-T62			2219-T81			2219-T87			"As Welded" 2219-T62, T81, T87			2014-T6 (T651)			"As Welded" 2014-T6 (T651)		
Temp ^{o}F	-423	-300	RT	-423	-300	RT	-423	-300	RT	-423	-300	RT	-423	-300	RT	-423	-300	RT
Ftu Ksi	85.3	64.8	54	94.2	74.4	62	91.3	75.6	63	36	33	30	89.1	75.2	66	41	37	34
Fty Ksi	48.9	41.9	36	61.1	54.2	46	69.1	60.8	52				74.2	66.1	58			
%	9.0	8.0	7.0	9.0	8.0	7.0	8.0	7.0	6.0				8.0	8.0	7.0			
Fcy Ksi	51.6	44	38	63.8	56.6	48	69.1	60.8	52				75.5	67	59			
Fsu Ksi	48	38	32	55	44	36	56	45	37				53.4	45.1	40			
E(10^6Psi)	12.3	11.7	10.5	12.3	11.7	10.5	12.3	11.7	10.5				12.2	11.7	10.5			
Ec(10^6Psi)	12.6	12.1	10.8	12.6	12.1	10.8	12.6	12.1	10.8				12.4	11.9	10.7			
N Ftu Ratio																		
Ftu (2) (K_t=]2)								0.87	0.91					0.78	0.93			
Density (1)			0.102			0.102			0.102						0.101			

NOTES: (1) Average Values

(2) Alcoa Data-for information only

TABLE 8.1-2. (Cont'd)
DESIGN ALLOWABLES - METALLIC MATERIALS

	Aluminum						Titanium						Magnesium					
Material	2021-T 87T			7178-T76(3)			5AL-2155M ELI			61A1-4V			LA141A-T7			LA91A (1)		
Temp	-423	-300	RT	-423	-300	RT	-423	-300	RT	-423	-300	RT	-423	-300	RT	-423	-300	RT
Ftu Ksi	85	79	66	-	87	74	210	165	105	-	230	160	36	24.8	18	46	36	22
Fty Ksi	75	69	58	-	75	64	190	150	95	-	208	150	24	15.1	12	35	24	16.5
e %	7.0	7.0	6.0	-	3.0	8.0	6.0	7.0	10.0	-	5.0	8.0	5.0	5.0	10	11	15	30
Fcy Ksi	75	69	59	-	76	65	190	150	95	-	216	150	26	16.3	13	37	23	19
Fsu Ksi	51	47	39	-	52	44	134	102	65	-	143	100	26	18	13	32	25	15
$E(10^6$ Psi)	12.4	11.9	10.7	-	11.0	10.3	17.0	16.9	15.5	-	17.6	16.0	-	-	6.1	-	-	6.6
$Ec(10^6$ Psi)	12.8	12.0	11.0	-	11.5	10.5	17.0	16.9	15.5	-	18.0	16.4						
N Ftu Ratio																		
Ftu $(K_t=12)^{(4)}$					0.34	0.70	0.83	1.15	1.29	-	0.58	1.00						
Density (2) (lbs/in^3)			0.103			0.102			0.162			0.160			0.0485			0.053

NOTES: (1) Typical Values
(2) Average Values
(3) Aluminum Alloy 7178 is considered unsatisfactory for Cryogenic Applications. Inadequate toughness characteristics in this temperature range.
(4) Alcoa data for information only.

Table 8.1-3 Advanced Composites - Preliminary Design Properties

Composite System / Fiber Volume % / Density lb in.^{3}		B/E W/Scrim ≃ 50 0.072				G/E (Type 1) ④ ≃ 55 0.056				G/E (Type -) ④ ≃ 55 0.056				S-Glass Scotchply Array ① 0.072				S-Glass Fila. Wound 0.072			
Property \ Temperature	Orientation	-423	-300	R.T.	350	-423	-300	R.T.	350	-423	-300	R.T.	350	-423	-300	R.T.	350	-423	-300	R.T.	350
F_{tu} (ksi)	[0]°	183	208	178	136	175	198	170	165	108	122	105	95	③	210	180	150	330		220	
	[±45]°	-	29	21	8	29	29	22	20	18	23	17	15								
	[90]°	6.6	8.4	8.6	5.4	3.8	4.9	5	3.5	3.6	4.7	4.8	3.0	③	5	5	3.5				
E_t (10^6 psi)	[0]°	-	32	30	25	-	25	19.6	18	-	34	29	28	③	6.5	6.0	5.5	9.2		8.4	
	[±45]°	③	3.6	3.5	1.3	③	③	3.0	2.2	③	≃ 3.2	2.8	2.5								
	[90]°	-	3.3	3.0	1.1	-	1.5	1.0	1.0	-	③	1.1	0.8	③	1.1	1.0	0.7				
% Ult. Tens. Strain	[0]°																	3.64		2.68	
F_{cu} (ksi)	[0]°	517	400	400	134	289	224	140	70	207	160	100	80		140	100	75	③	240	165	
	[±45]°	-	39	28	22	-	43	35	31	-	25	18						③	28	15	
	[90]°	③	76	32	15	③	73	35	25	③	73	35	25					③	60	30	
E_c (10^6 psi)	[0]°	30	33	28	28	-	25	19.6	19			27	25					③	9.0	7.5	
	[±45]°	-	4.0	3.2	2.8	-	3.0	2.4	2			2.6						③	3.0	2.2	
	[90]°	③	5.0	2.8	1.5	③	1.5	1.0	0.9			1.5	1.0					③	2.8	1.8	
F_{su} (ksi)	[0]°	-	18	8.5	3.0	③	15	7.0	6.5	5.5	5.8	5.0	4.7					9	8	6	
	[±45]°	③	80	61	30	50	55	44	40	30	33	25	22					42	40	35	Very extensively with resin system
In-Plane Shear	[90]°	-	16	8.5	2.9	③	15	7.0	6.5	5.5	5.8	5.0	4.7					9	8	6	
G_s (typ.)	[0]°	-	1.5	1.0	0.6	-	1.2	0.81	0.75	-	1.2	0.81	0.75					1.0	0.95	0.8	
	[±45]°	③	9.0	8.5	8.6	③	6.0	5.25	5.00	③	8.5	7.6	7.0					2.5	2.4	2.2	
(10^6 x psi)	[90]°	-	1.4	1.0	0.6	-	1.2	0.81	0.75	-	1.2	0.81	0.75					1.0	0.95	0.8	
Poisson's Ratio	[0]°			.21	.20			.21	.21			.30	.30								
	[±45]°			.45	.92			.80	.88			.83	.84								
	[90]°			.02	.01			.02	.01			.02	.01								
α (μ in./in.-F°)	[0]°	-	2.3	2.0	3.0	≃ 0	≃ 0	≃ 0	≃ 0	≃ 0	≃ 0	≃ 0	≃ 0								
	[90]°	③	6.0	10.8	20	-	13	17	13		13	17	13								
	⊥		②	②	②			②	25			②	25								
K (Btu-in./hr-ft^2-F°)	[0]°	≃ 13	13.2	14.2	14	≃17.0	25.5	75	110		115.0	375							3.0	5.7	
	[90]°	≃ 1.5	1.75	2.7	3.0	-	≃4.0	4.5	6.0	-	5.0	7.5						- ⑤	≃1.7	≃3.2	

① Values based on proportional relationship between Boron/Epoxy and S-Glass/Epoxy values in Mil. Handbook 17.

② These coefficients are instantaneous values - other values are average coefficients

③ Where -423 F values are not shown, use -300 values; where -300 values are not shown, use R.T. values

④ Type 1 is Modmor II or HTS; Type 4 is Modmore I or HMS Fiber

⑤ G.D.Report ML/TRD 64-33 Pt. III

<u>NOTE:</u> All design values are predicated on very limited test data and are subject to change. Most cryogenic mechanical property values are based on engineering judgement, influenced by past experience with glass fiber reinforced composites. All are conservative estimates of 1972 technology capability. Improvement of ≃ 20% should be realized by 1976.

TABLE 8.2-1

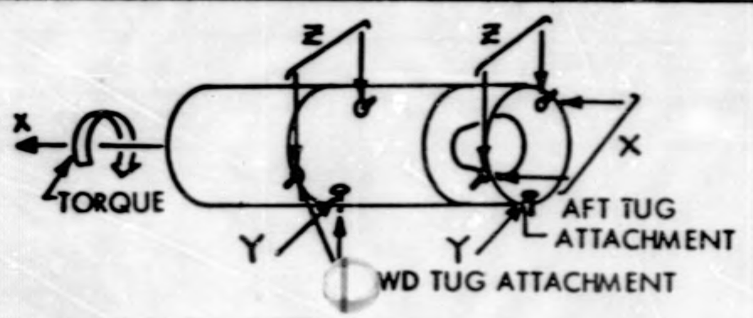
SHUTTLE PAYLOAD LOAD FACTORS CONDITIONS

Condition	X(g)	Y(g)	Z(g)
Launch	1.4 ± 1.6	± 1.0	± 1.0
High Q Booster Thrust	1.9 ± 0.3	± 1.0	0.8 ± 0.2
End Boost (Booster Thurst)*	3 ± 0.3	± 0.6	± 0.6
End Burn (Orbiter Thrust)	3 ± 0.3	± 0.5	± 0.5
Orbiter Entry	-0.5	± 1.0	-3.0 ± 1.0
Orbiter Flyback	-0.5	± 1.0	± 1.0 -2.5 ± 1.0
Landing	-1.3	± 0.5	-2.7 ± 0.5

*Excludes booster-orbiter separation loads which are TBD.

Table 8.2-2 Space Tug Attachment Limit Loads

CONDITION	TUG WT (KIPS)	PAYLOAD WT (KIPS)	TUG ACCELERATIONS (G'S)			ATTACHMENT LOADS					
						FWD TUG (AFT EOS)			AFT TUG (FWD EOS)		
			X	Y	Z	Y (KIPS)	Z (KIPS)	TORQUE (IN.LB X 10-6)	X (KIPS)	Y (KIPS)	Z (KIPS)
LAUNCH	56.94	8.06	-3.0	±1.0	±1.0	± 35	± 35	∓ 3.20	-195	± 30	± 30
LAUNCH	56.94	8.06	+0.2	±1.0	±1.0	± 35	± 35	∓ 3.20	+ 13	± 30	± 30
END BOOST	56.94	8.06	-3.3	±0.6	±0.6	± 21	± 21	∓ 1.92	-214	± 18	± 18
LANDING	6.36	4.16	+ 1.3	±0.5	+ 3.2	± 4	+ 29	∓ 0.41	+ 14	± 1	+ 5
ORBITER ENTRY	6.36	4.16	+ 0.5	±1.0	+ 4.0	± 9	± 36	∓ 0.82	+ 5	± 1	+ 6
LAUNCH	62.00	0	- 3.0	±1.0	±1.0	± 26	± 26	∓ 2.35	-186	±36	± 36
LAUNCH	62.00	0	+ 0.2	±1.0	±1.0	± 26	± 26	∓ 2.35	+ 12	±36	± 36
END BOOST	62.00	0	- 3.3	±0.6	± 0.6	± 16	± 16	∓ 1.41	-205	± 22	± 22



CONCENTRATED WEIGHTS & TOTAL WT

	WT (LB)	CG STATION	LOAD STATION
3K PAYLOAD + TUG	65000	229.56	—
ENGINE	288	90.2	156.0
LOX + TANK + T/S	48345	179.83	156.0
LH_2 + TANK	8589	357.06	313.5
4.16K PAYLOAD	10517	395.05	—
ENGINE	288	90.2	156.0
LOX TANK + T/S	871	166.2	156.0
LH_2 TANK	420	348.1	313.5
8.06 K PAYLOAD	65000	264.3	—
ENGINE	288	90.2	156.0
LOX + TANK + T/S	44008	183.80	156.0
LH_2 + TANK	7866	364.22	313.5

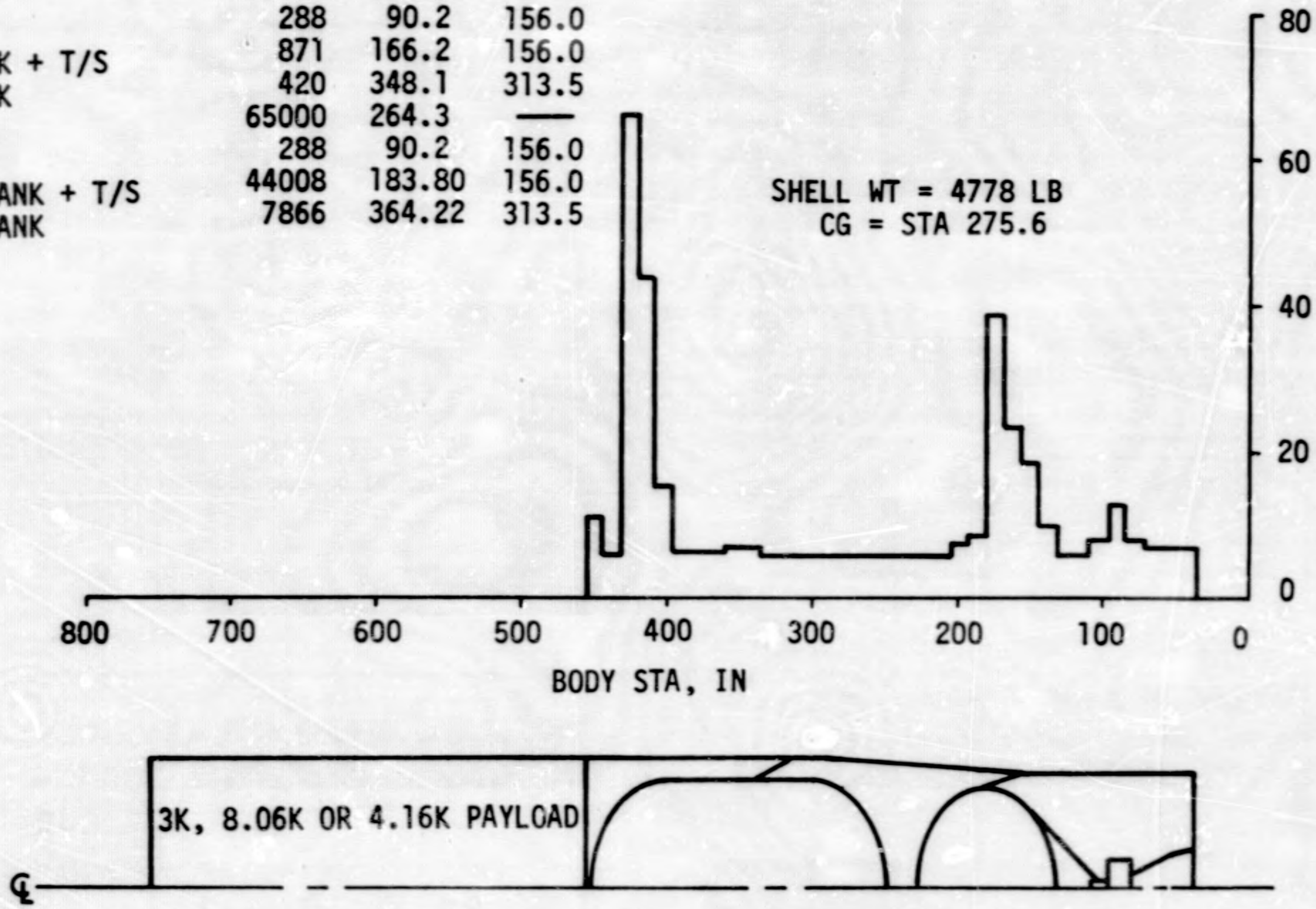


Figure 8.2-1 Space Tug Weight Distribution

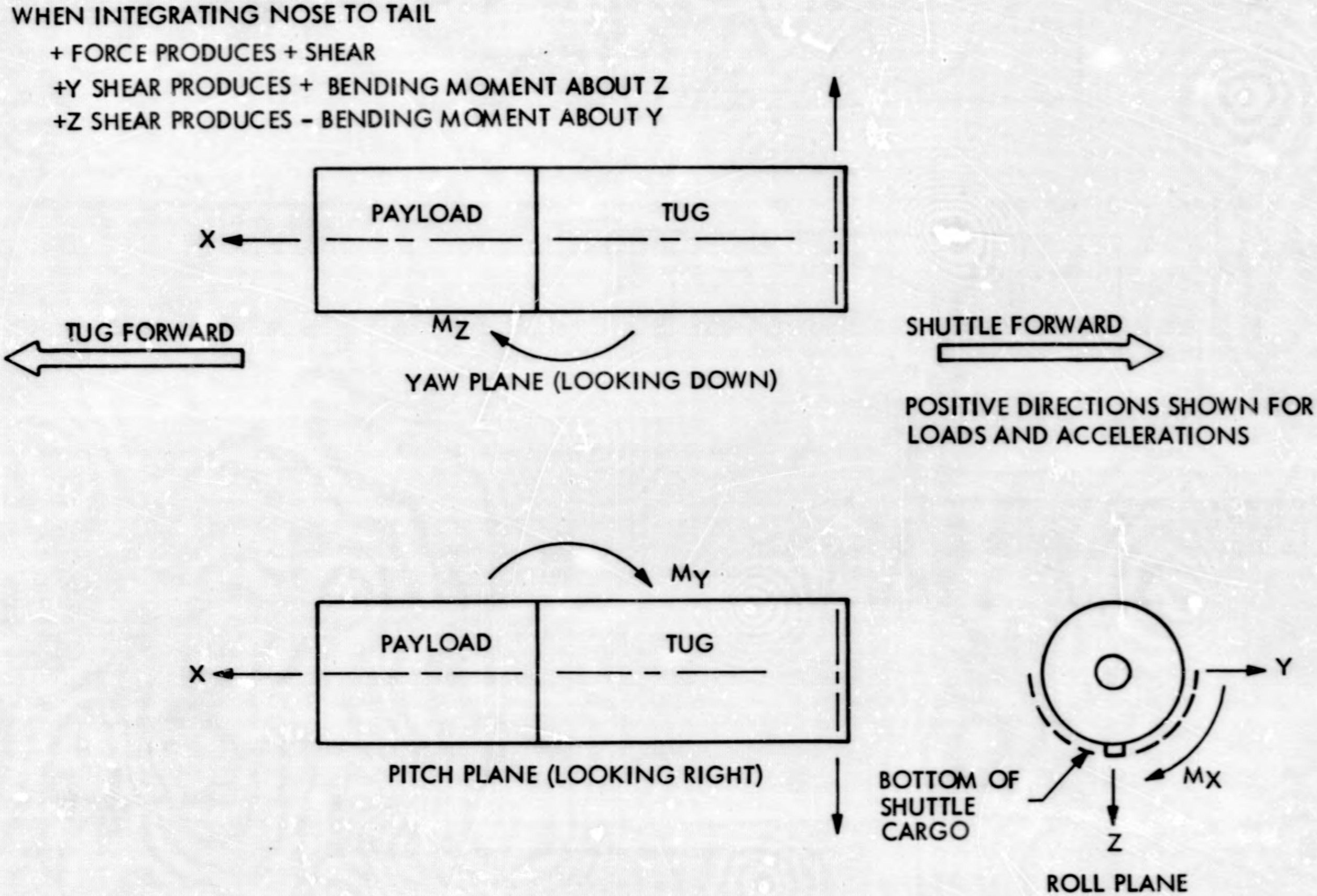


Figure 8.2-2 Loads Sign Convention

Figures 8.2-3 through 8.2-18 present the critical Space Tug body loads. All loads were calculated using a rigid body analysis.

8.2.2 Environment

The Tug was designed to meet the following environmental characteristics:

a) Thermal

The Tug shell thermal environment is assumed to be that of the orbiter cargo bay and is defined in Table 8.2-3. Detail design temperature for the Tug tank structure is found in Section 8.4 of this report.

b) Vibrations

The Tug environment for vehicle dynamics, lift-off random vibrations, boost random vibrations, and shock spectrum have not been defined for this study.

c) Acoustic

The Tug acoustic environment is assumed to be that of the orbiter cargo bay and is defined in Table 8.2-4.

d) Payload Bay Atmosphere

The orbiter payload bay is capable of atmospheric control independent of the orbiter internal structure while on the launch pad. This provision allows the control temperature, humidity, and atmospheric composition by the payload. The GSE connections for the payload conditioning shall be vented during launch and entry phases and operate unpressurized during the orbital phase of the mission. Payload bay pressure is assumed equal to the orbiter ambient pressure during all operational phases.

8.3 UNPRESSURIZED STRUCTURAL SUBSYSTEM

The design description and analysis of the Space Tug unpressurized structure is presented in this section. The presentation is divided into three major subsections; the body shell structure, the Tug-to-Space Shuttle Orbiter adapter structure, and the thrust structure. The body shell structure is 322 inches long consisting of two cylinders and a conical frustrum. Included are the Tug aft skirt, intertank structure, and forward skirt. All three of these assemblies are of honeycomb sandwich construction, utilizing aluminum core and face sheets of advanced composites (primarily graphite-epoxy). In addition to reacting body loads the shell structure provides micro-meteoroid protection and backs up the insulation purge bag system. Design features that are common to the three shells are delineated in the shell structure general description and not repeated in the detail discussion of each assembly. The Tug-to-Shuttle adapter is a cylinder 81 inches in length. Its primary function is acceptance of concentrated loads at the Shuttle interface and converting them to near uniform loading at the Tug aft skirt interface. The adapter construction is similar to the body shell with the addition of tapered titanium

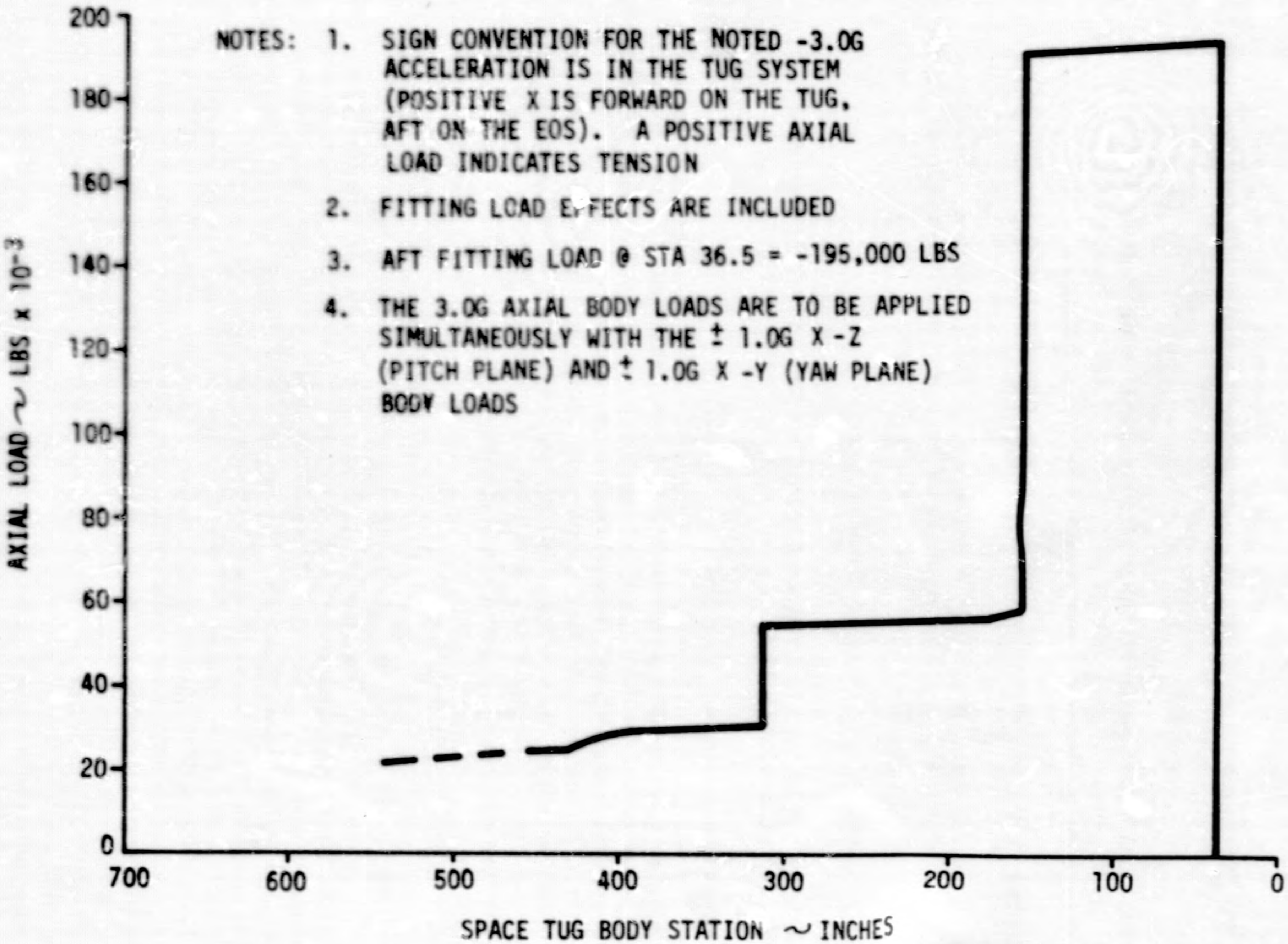


Figure 8.2-3 Space Tug + 8060 lb Payload in Space Shuttle Cargo Bay -3.0G Limit Axial Body Loads

NOTES: 1. SIGN CONVENTION FOR THE NOTED +0.2G ACCELERATION IS IN THE TUG SYSTEM (POSITIVE X IS FORWARD ON THE TUG, AFT ON THE EOS) A NEGATIVE AXIAL LOAD INDICATES COMPRESSION

2. FITTING LOAD EFFECTS ARE INCLUDED

3. AFT FITTING LOAD @ STA 36.5 = 13.000 LBS

4. THE -0.2G AXIAL BODY LOADS ARE TO BE APPLIED SIMULTANEOUSLY WITH THE ± 1.0G X -Z (PITCH PLANE) AND ± 1.0G X -Y (YAW PLANE) BODY LOADS

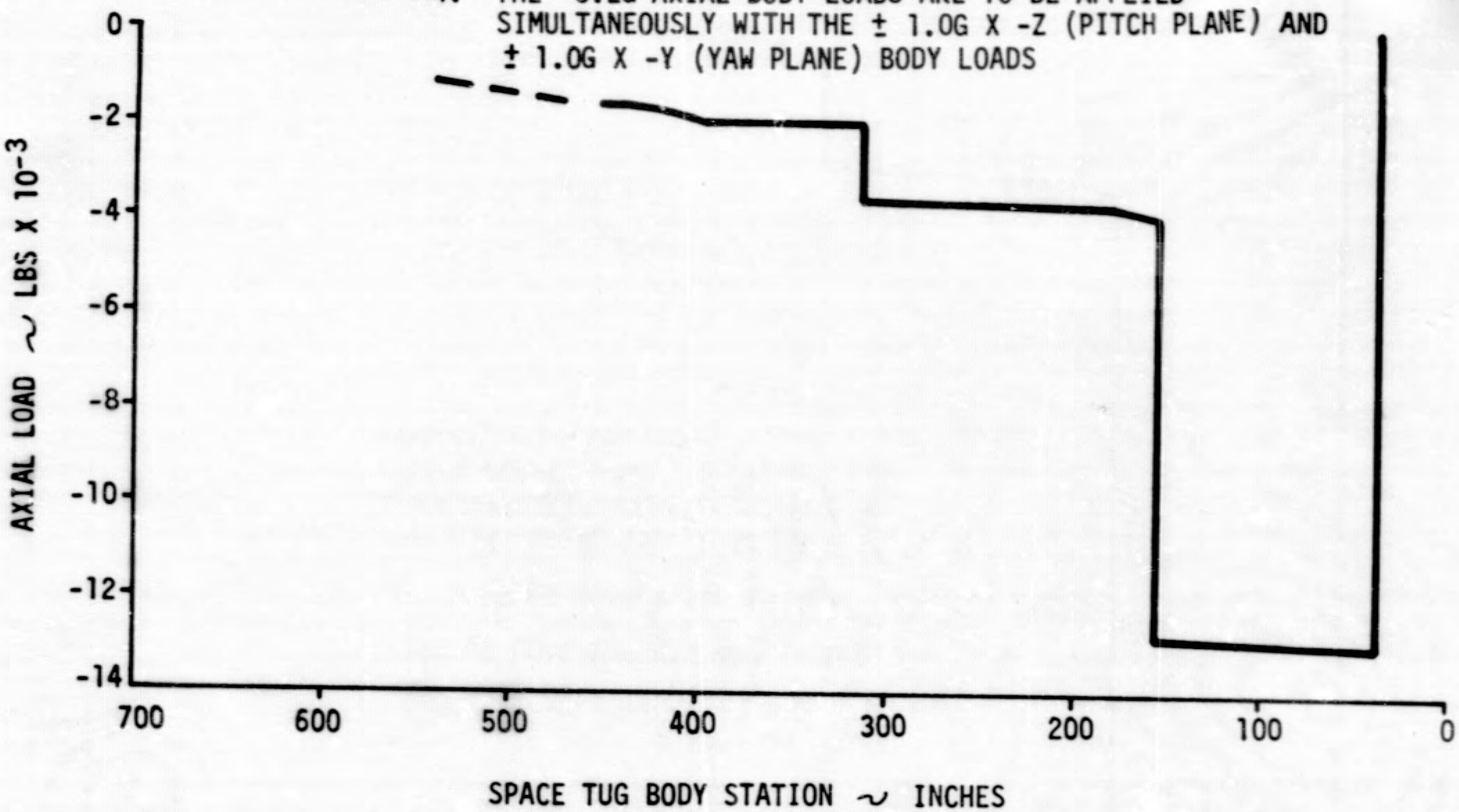


Figure 8.2-4 Space Tug + 8060 Lb Payload in Space Shuttle Cargo Bay + 0.2G Limit Axial Body Loads

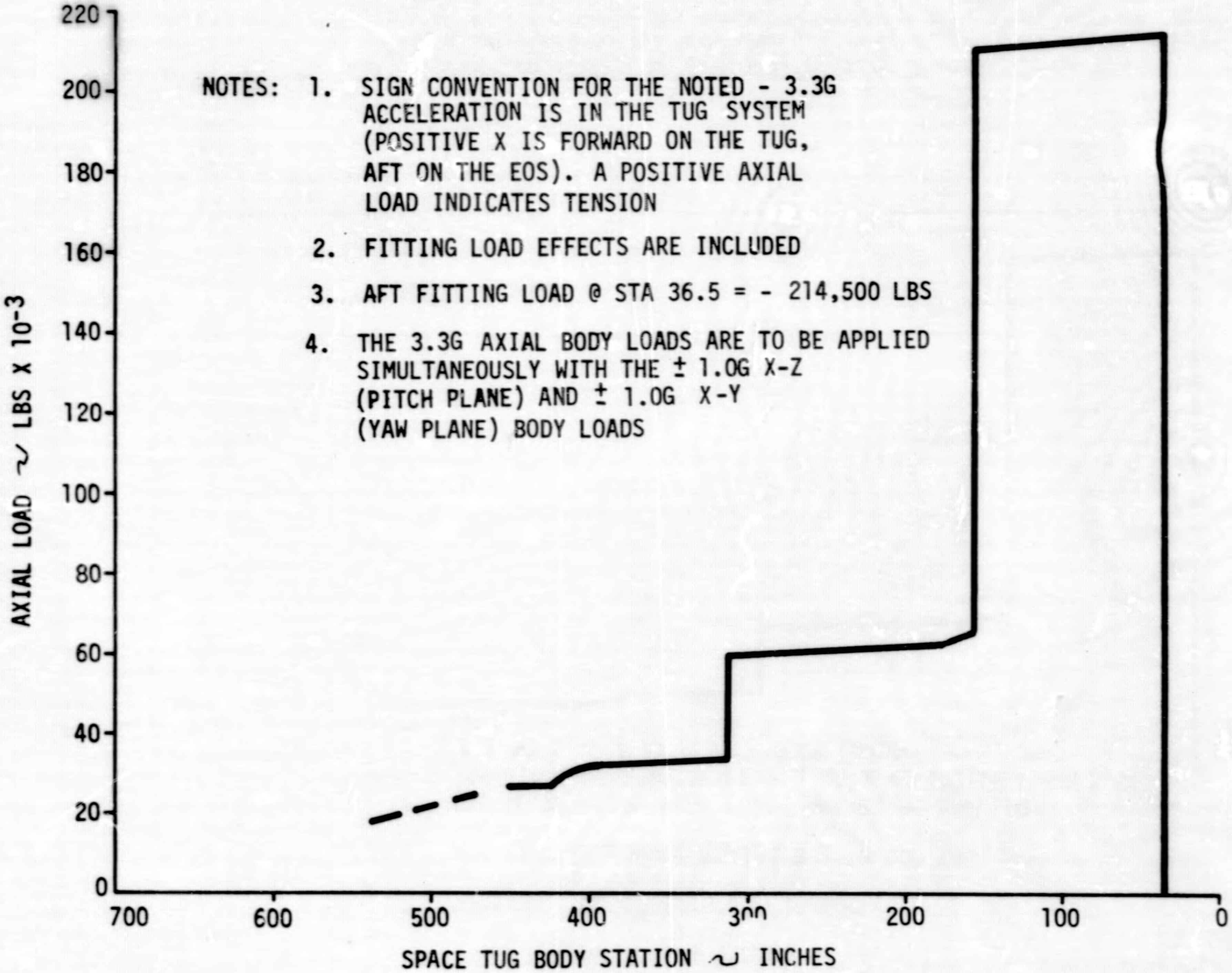


Figure 8.2-5 Space Tug + 8060 Lb Payload in Space Shuttle Cargo Bay -3.3G Limit Axial Body Loads

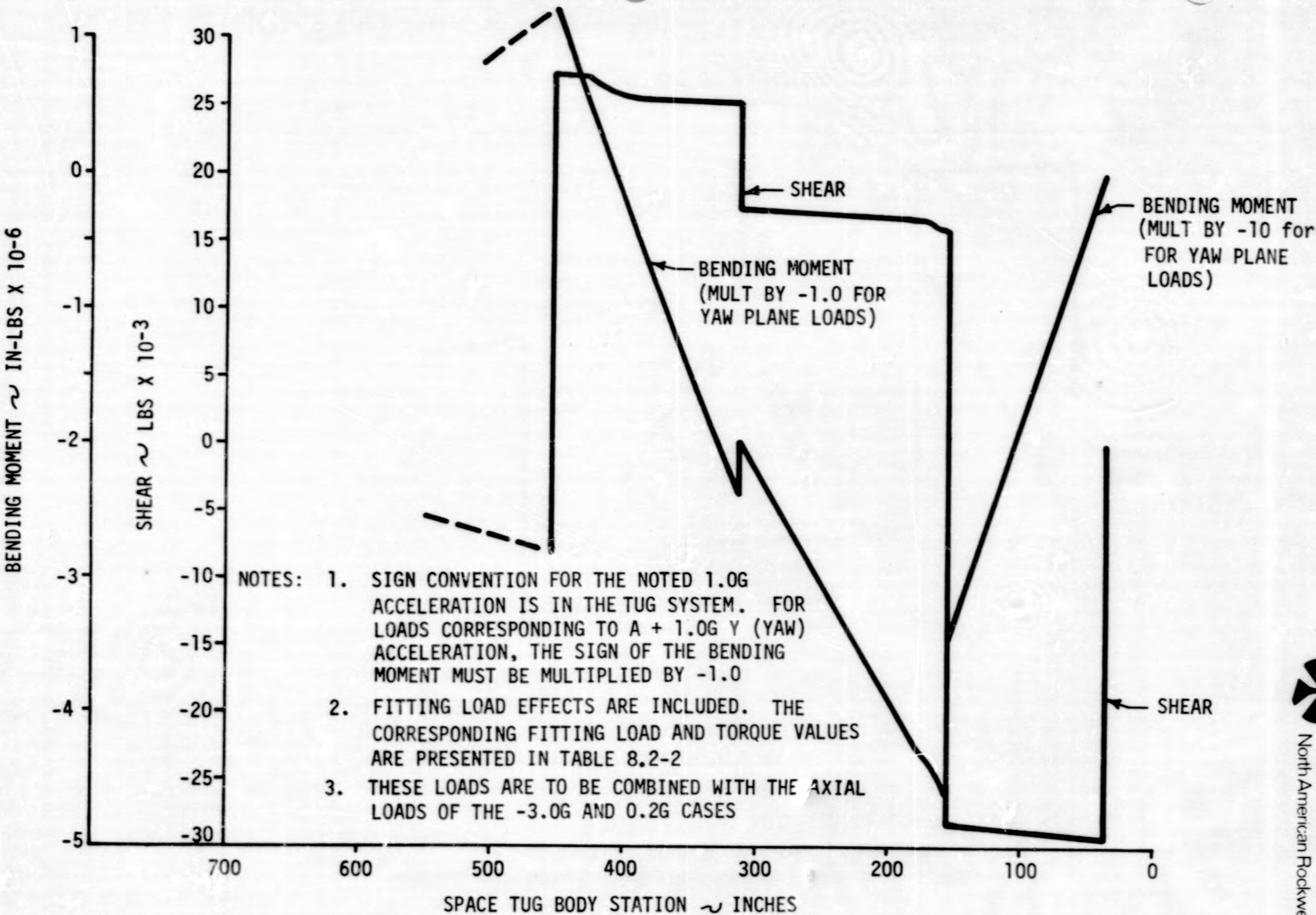


Figure 8.2-6 Space Tug + 8060 lb Payload in Space Shuttle Cargo Bay
1.0G X-Z and X-Y Plane Limit Body Loads

NOTES: 1. SIGN CONVENTION FOR THE NOTED 0.6G ACCELERATION IS IN THE TUG SYSTEM. FOR LOADS CORRESPONDING TO A + 0.6G Y (YAW) ACCELERATION, THE SIGN OF THE BENDING MOMENT MUST BE MULTIPLIED BY -1.0

2. FITTING LOAD EFFECTS ARE INCLUDED. THE CORRESPONDING FITTING LOAD AND TORQUE VALUES ARE PRESENTED IN TABLE 8.2-2

3. THESE LOADS ARE TO BE COMBINED WITH THE AXIAL LOADS OF THE -3.3G CASE

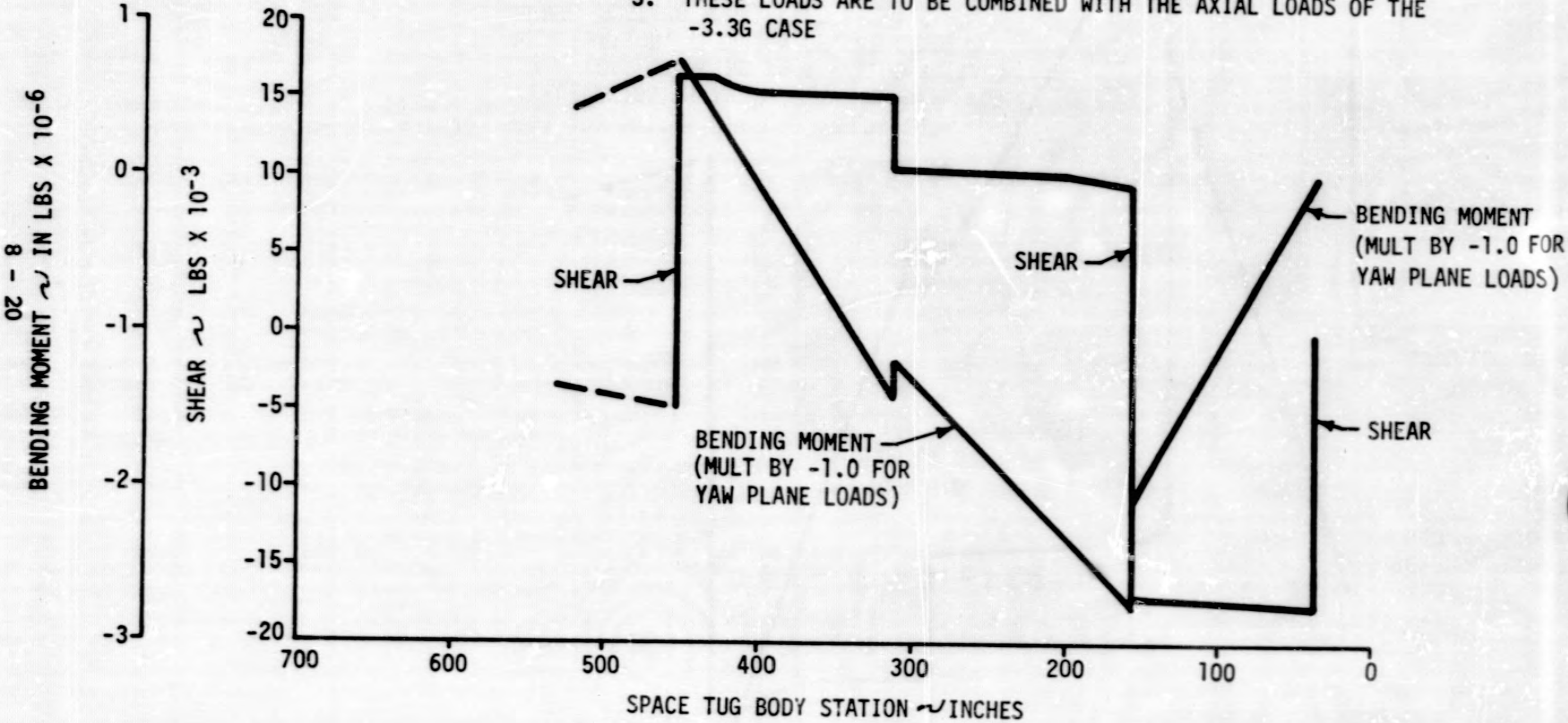


Figure 8.2-7 Space Tug + 8060 Lb Payload in Space Shuttle Cargo Bay 0.6G X-Z and X-Y Plane Limit Body Loads

NOTES: 1. SIGN CONVENTION FOR THE NOTED + 1.3G ACCELERATION IS IN THE TUG SYSTEM (POSITIVE X IS FORWARD ON THE TUG, AFT ON THE EOS). A NEGATIVE AXIAL LOAD INDICATES COMPRESSION

2. FITTING LOAD EFFECTS ARE INCLUDED

3. AFT FITTING LOAD @ STA. 36.5 = 13,670 LBS

4. THE 1.3G AXIAL BODY LOADS ARE TO BE APPLIED SIMULTANEOUSLY WITH THE +3.2G X-Z (PITCH PLANE) AND ± 0.5G X-Y (YAW PLANE) BODY LOADS

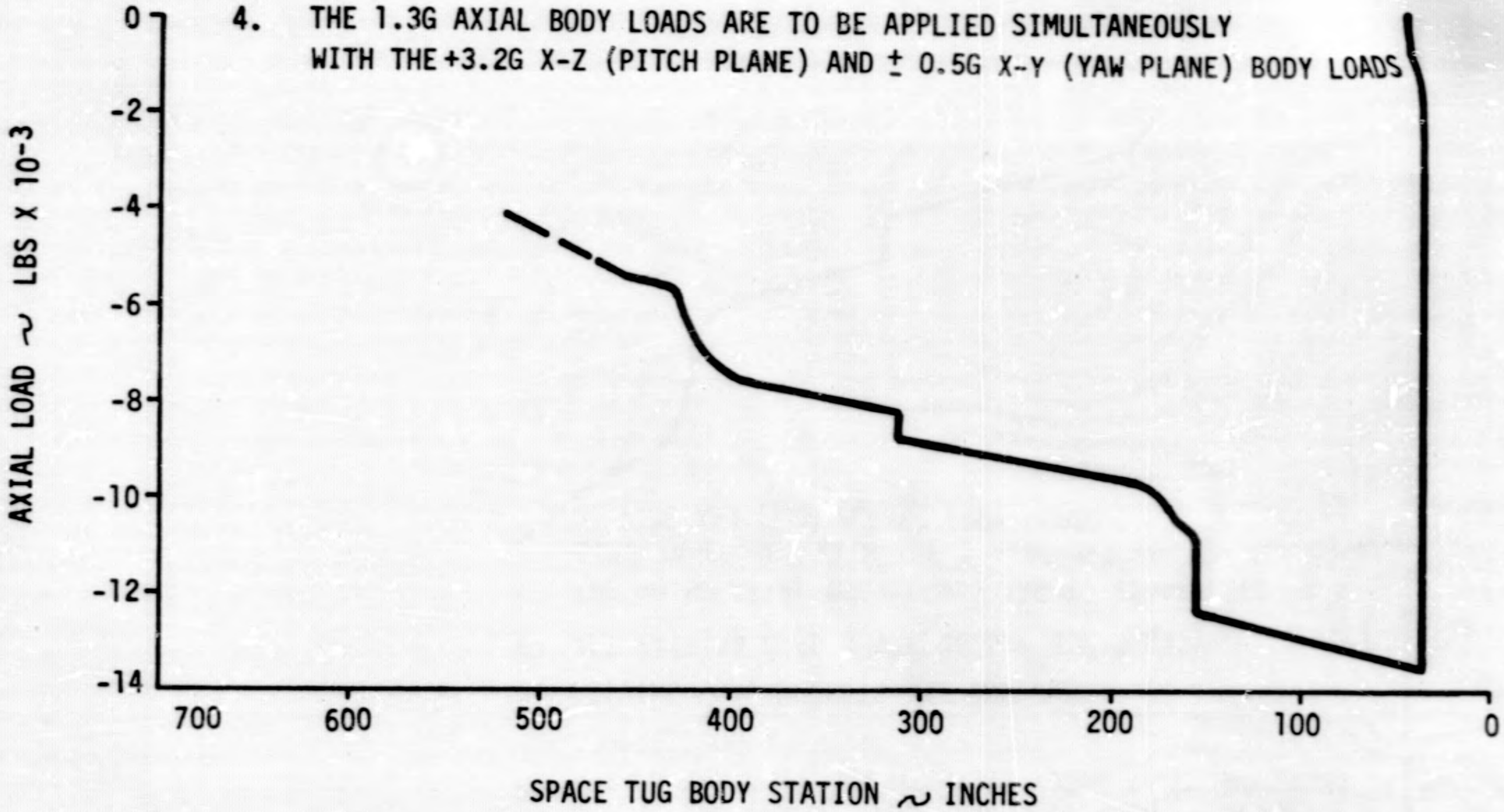


Figure 8.2-8 Space Tug (Empty) + 4160 lb Payload in Space Shuttle Cargo Bay 1.3G Limit Axial Body Loads

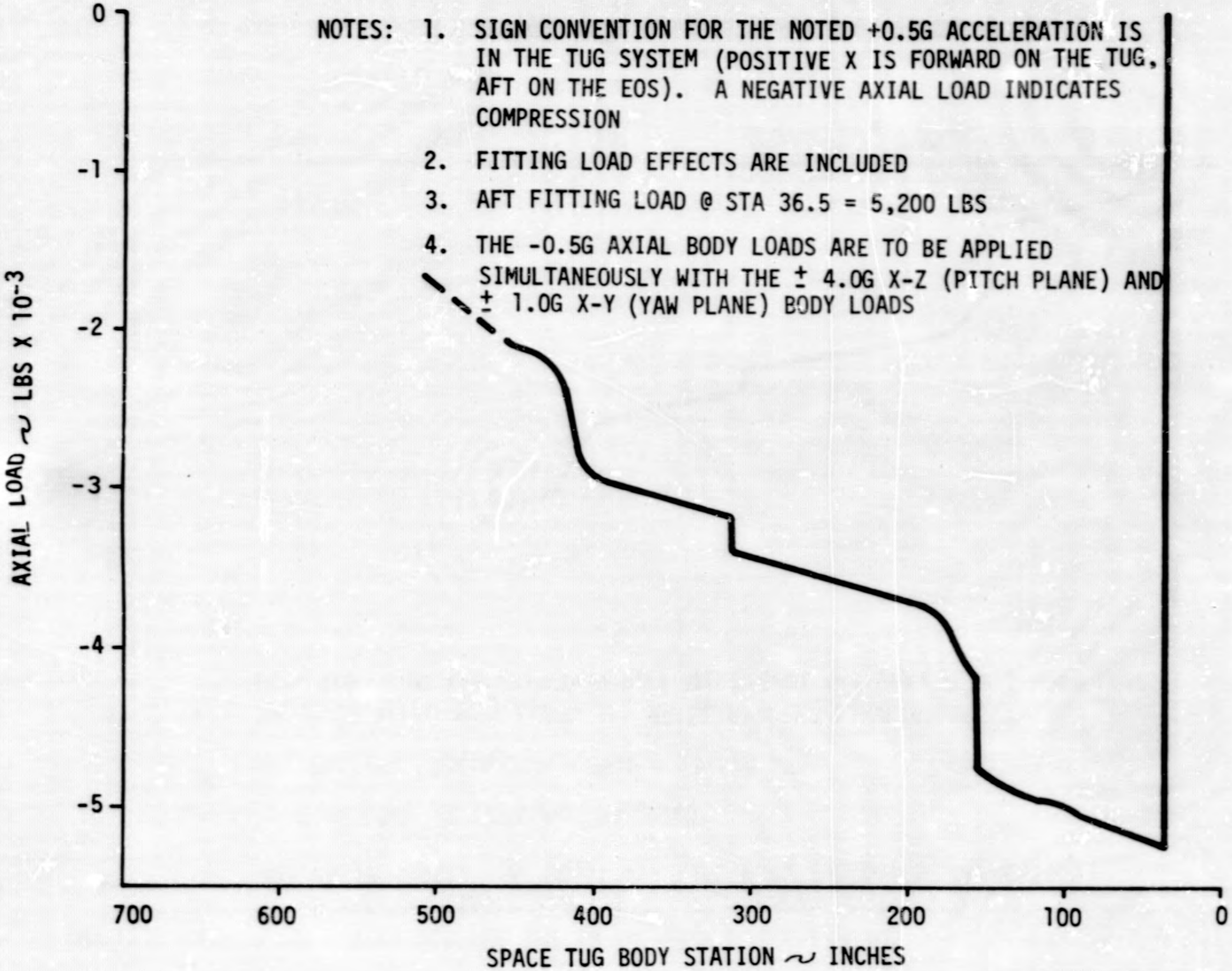


Figure 8.2-9 Space Tug (Empty) + 4160 lb Payload in Space Shuttle Cargo Bay + 0.5G Limit Axial Body Loads

NOTES:
1. SIGN CONVENTION FOR THE NOTED 1.0G ACCELERATION IS IN THE TUG SYSTEM. FOR LOADS CORRESPONDING TO A -1.0G Y (YAW) ACCELERATION, THE SIGN OF THE BENDING MOMENT AND SHEAR MUST BE MULTIPLIED BY -1.0
2. FITTING LOAD EFFECTS ARE INCLUDED. THE CORRESPONDING FITTING LOAD AND TORQUE VALUES ARE PRESENTED IN TABLE 8.2-2
3. THESE LOADS ARE TO BE COMBINED WITH THE AXIAL LOADS OF THE + 0.5G CASE AND THE + 4.0G PITCH PLANE LOADS

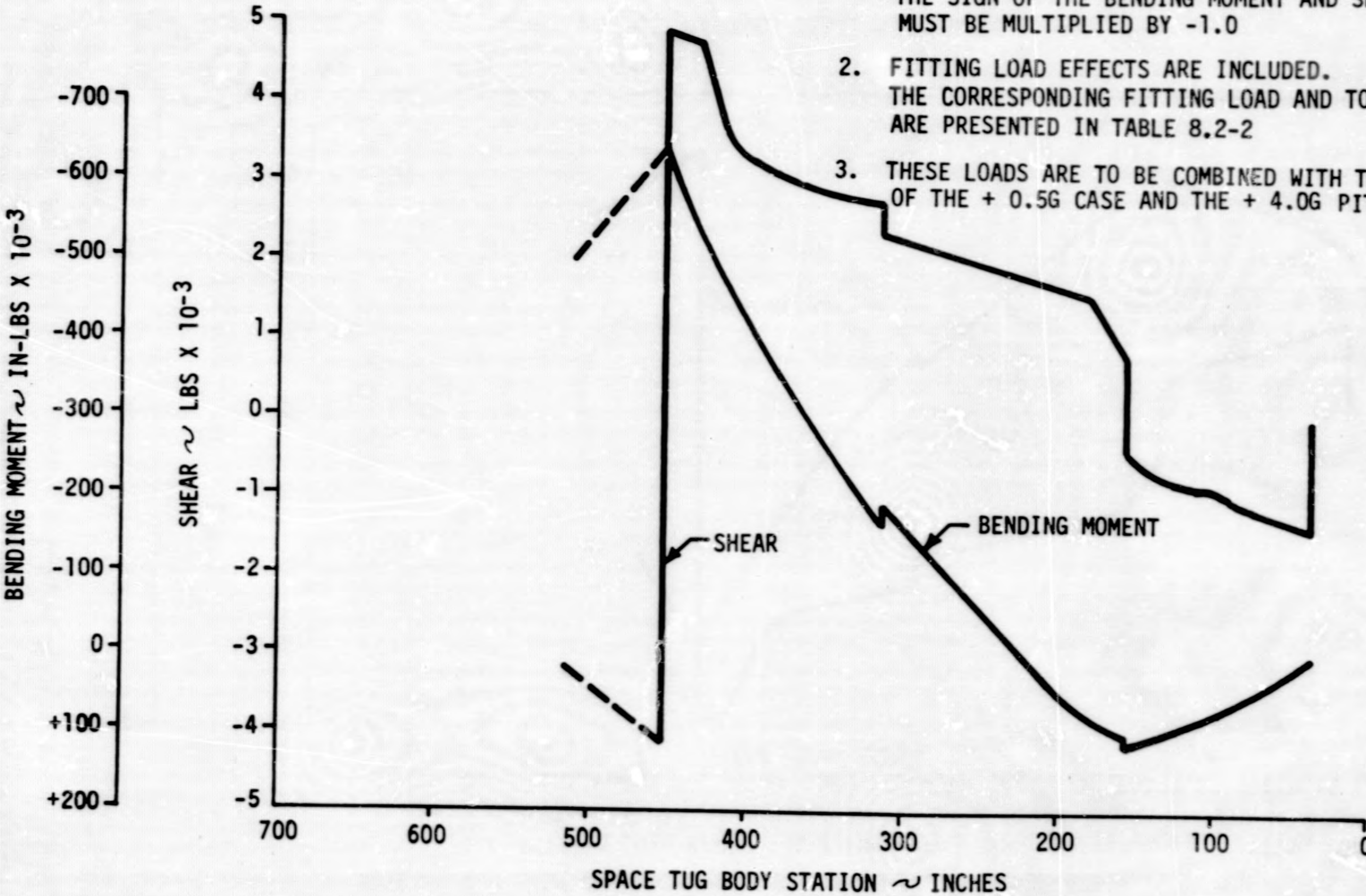


Figure 8.2-10 Space Tug (Empty) + 4160 Lb Payload in Space Shuttle Cargo Bay 1.0G X-Y Plane Limit Body Loads

NOTES: 1. SIGN CONVENTION FOR THE NOTED 4.0G ACCELERATION IS IN THE TUG SYSTEM

2. FITTING LOAD EFFECTS ARE INCLUDED. THE CORRESPONDING FITTING LOAD AND TORQUE VALUES ARE PRESENTED IN TABLE 8.2-2

3. THESE LOADS ARE TO BE COMBINED WITH THE AXIAL LOADS OF THE 0.5G CASE AND THE ± 1.0G YAW PLANE LOADS

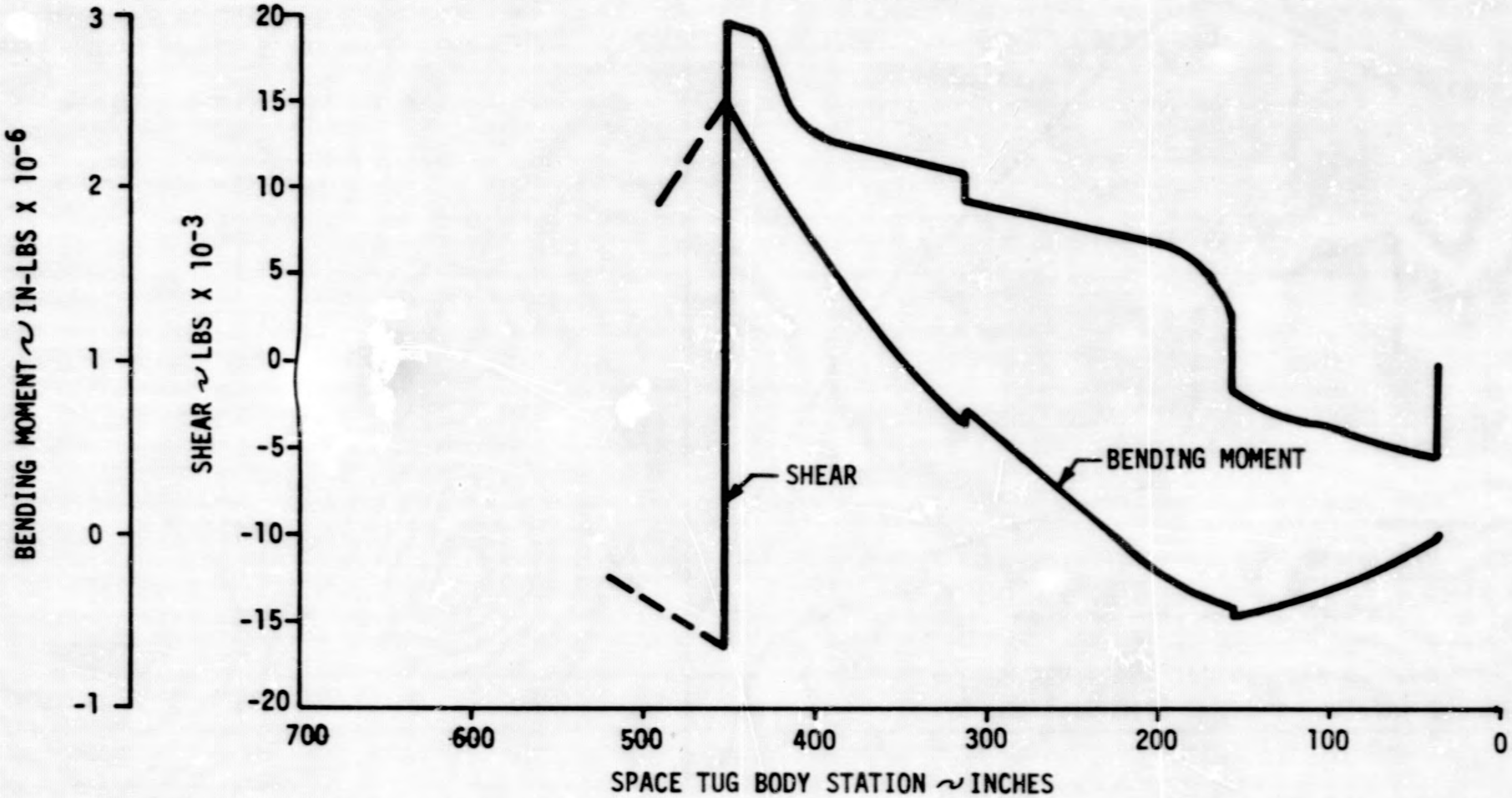


Figure 8.2-11 Space Tug (Empty) + 4160 lb Payload in Space Shuttle Cargo Bay 4.0G X-Z Plane Limit Body Loads

NOTES: 1. SIGN CONVENTION FOR THE NOTED 3.2G ACCELERATION IS IN THE TUG SYSTEM

2. FITTING LOAD EFFECTS ARE INCLUDED. THE CORRESPONDING FITTING LOAD AND TORQUE VALUES ARE PRESENTED IN TABLE 8.2-2

3. THESE LOADS ARE TO BE COMBINED WITH THE AXIAL LOADS OF THE 1.3G CASE AND THE ± 0.5G YAW PLANE LOADS

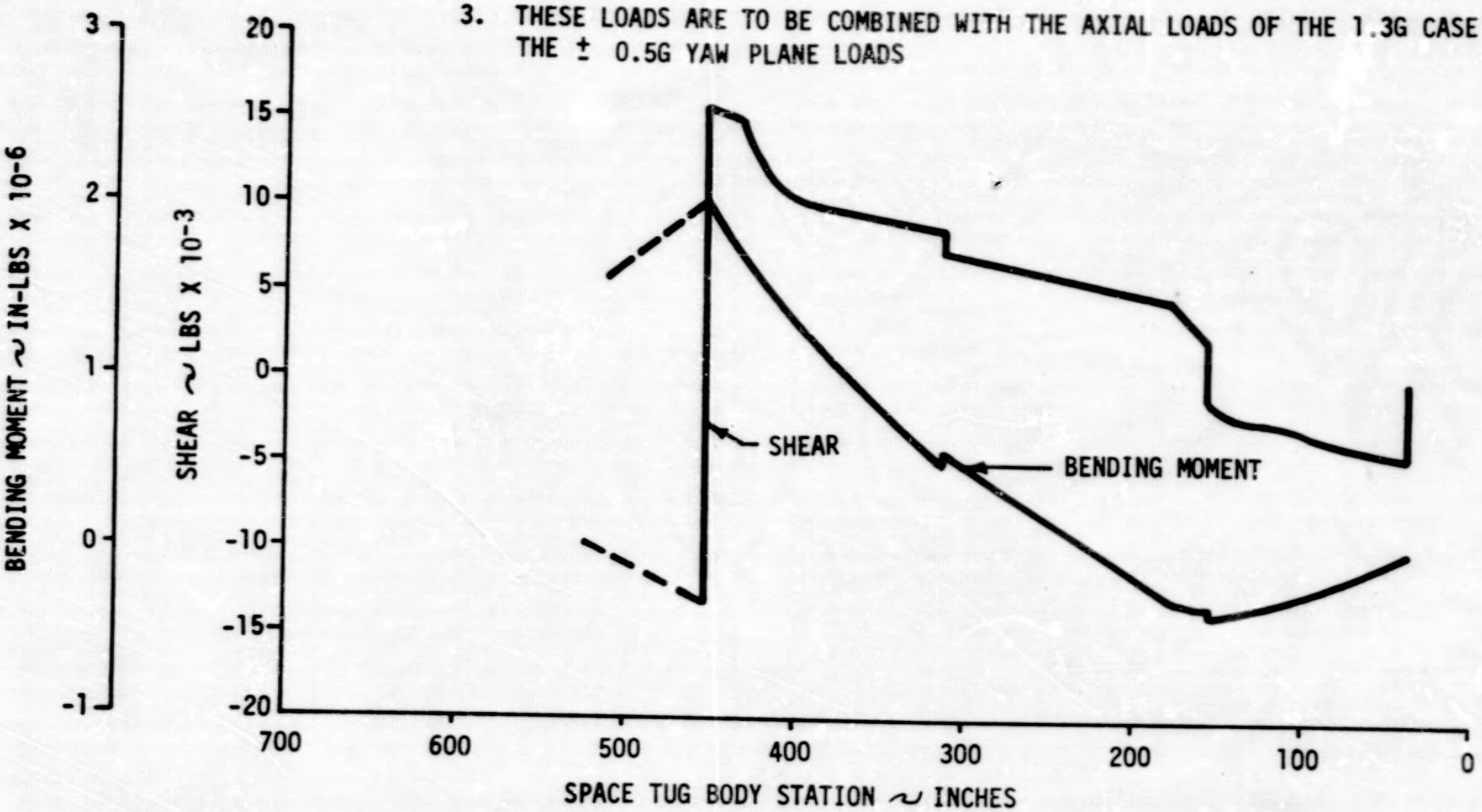


Figure 8.2-12 Space Tug (Empty) + 4160 lb Payload in Space Shuttle Cargo Bay 3.2G X-Z Plane Limit Body Loads

Space Division
North American Rockwell

NOTES: 1. SIGN CONVENTION FOR THE NOTED 0.5G ACCELERATION IS IN THE TUG SYSTEM. FOR LOADS CORRESPONDING TO A 0.5G Y (YAW) ACCELERATION, THE SIGN OF THE BENDING MOMENT AND SHEAR MUST BE MULTIPLIED BY -1.0

2. FITTING LOAD EFFECTS ARE INCLUDED. THE CORRESPONDING FITTING LOAD AND TORQUE VALUES ARE PRESENTED IN TABLE 8.2-2

3. THESE LOADS ARE TO BE COMBIND COMBINED WITH THE AXIAL LOADS OF THE + 1.3G CASE AND THE + 3.2G PITCH PLANE LOADS

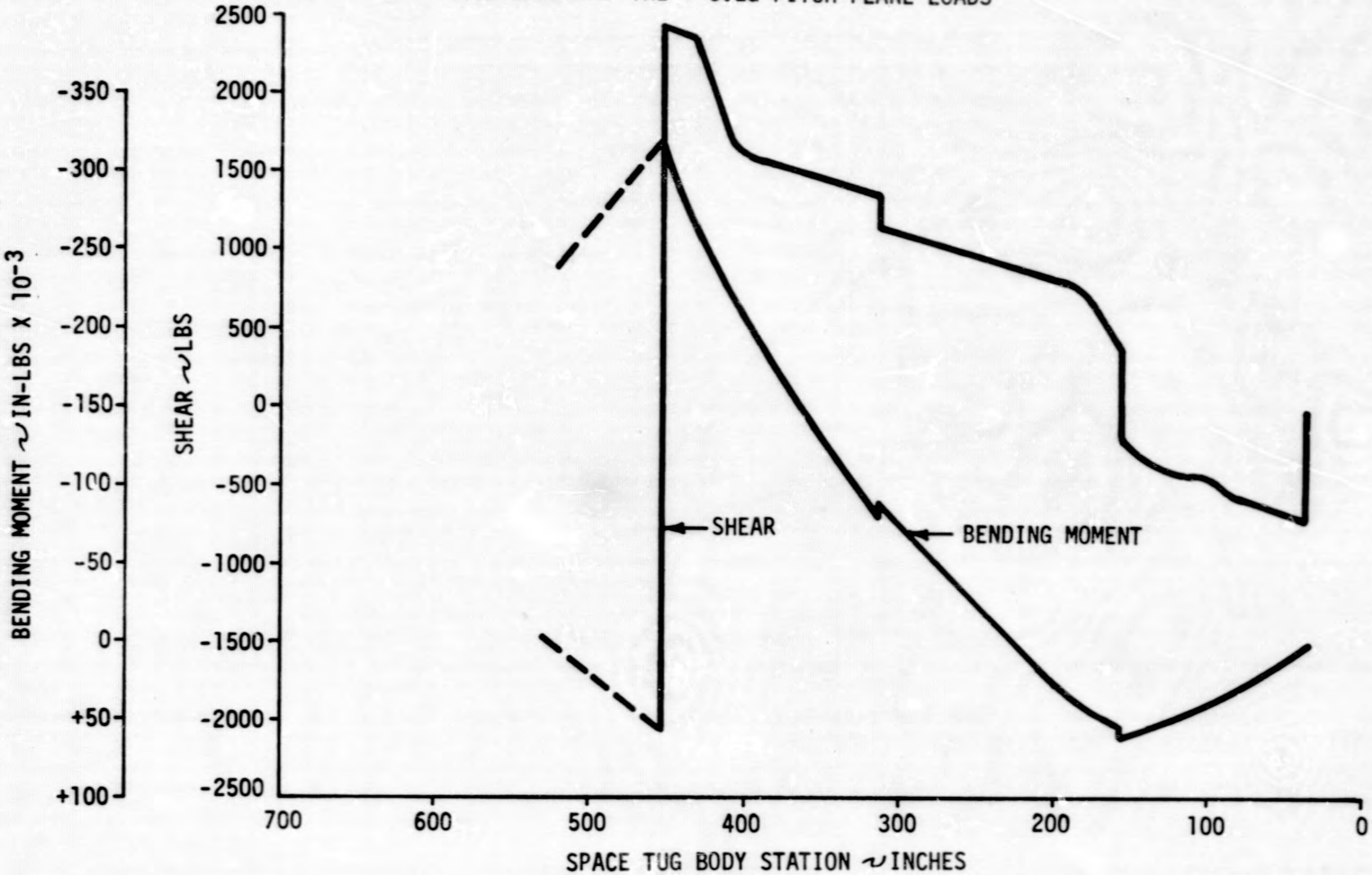


Figure 8.2-13 Space Tug (Empty) + 4160 lb Payload in Space Shuttle Cargo Bay 0.5G X-Y Plane Limit Body Loads

Space Division
North American Rockwell

NOTES:
1. SIGN CONVENTION FOR THE NOTED 0.2G ACCELERATION IS IN THE TUG SYSTEM (POSITIVE X IS FORWARD ON THE TUG, AFT ON THE EOS). A NEGATIVE AXIAL LOAD INDICATES COMPRESSION
2. FITTING LOAD EFFECTS ARE INCLUDED
3. AFT FITTING LOAD AT STA 36.5 = 12,400 LBS
4. THESE LOADS ARE TO BE APPLIED SIMULTANEOUSLY WITH THE $\pm$ 1.0G PITCH AND $\pm$ 1.0G YAW PLANE LOADS

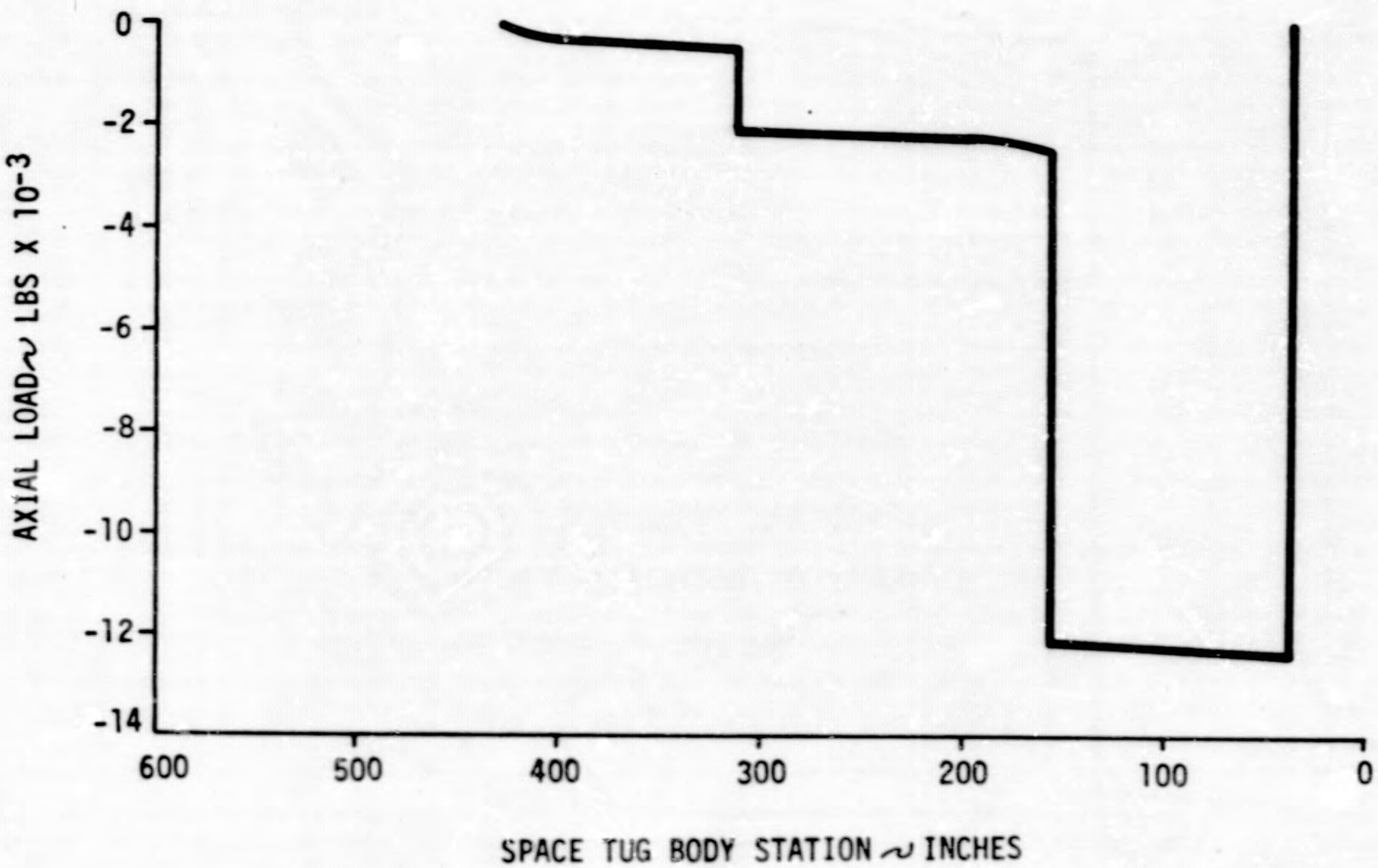


Figure 8.2-14 Space Tug (62K, No Payload) in Space Shuttle Cargo Bay 0.2G Limit Axial Body Loads

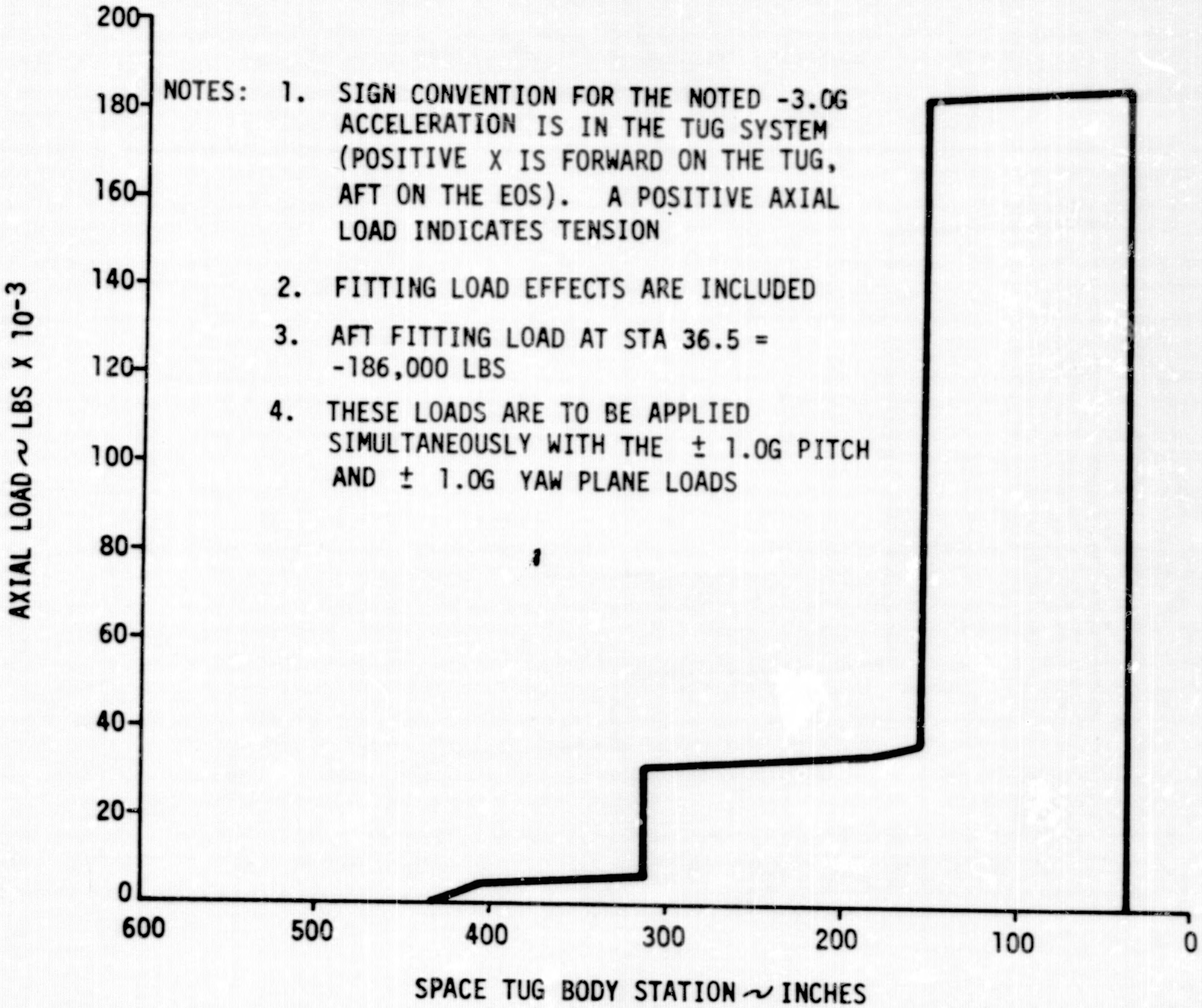


Figure 8.2-15 Space Tug (62K, No Payload) in Space Shuttle Cargo Bay -3.0G Limit Axial Body Loads

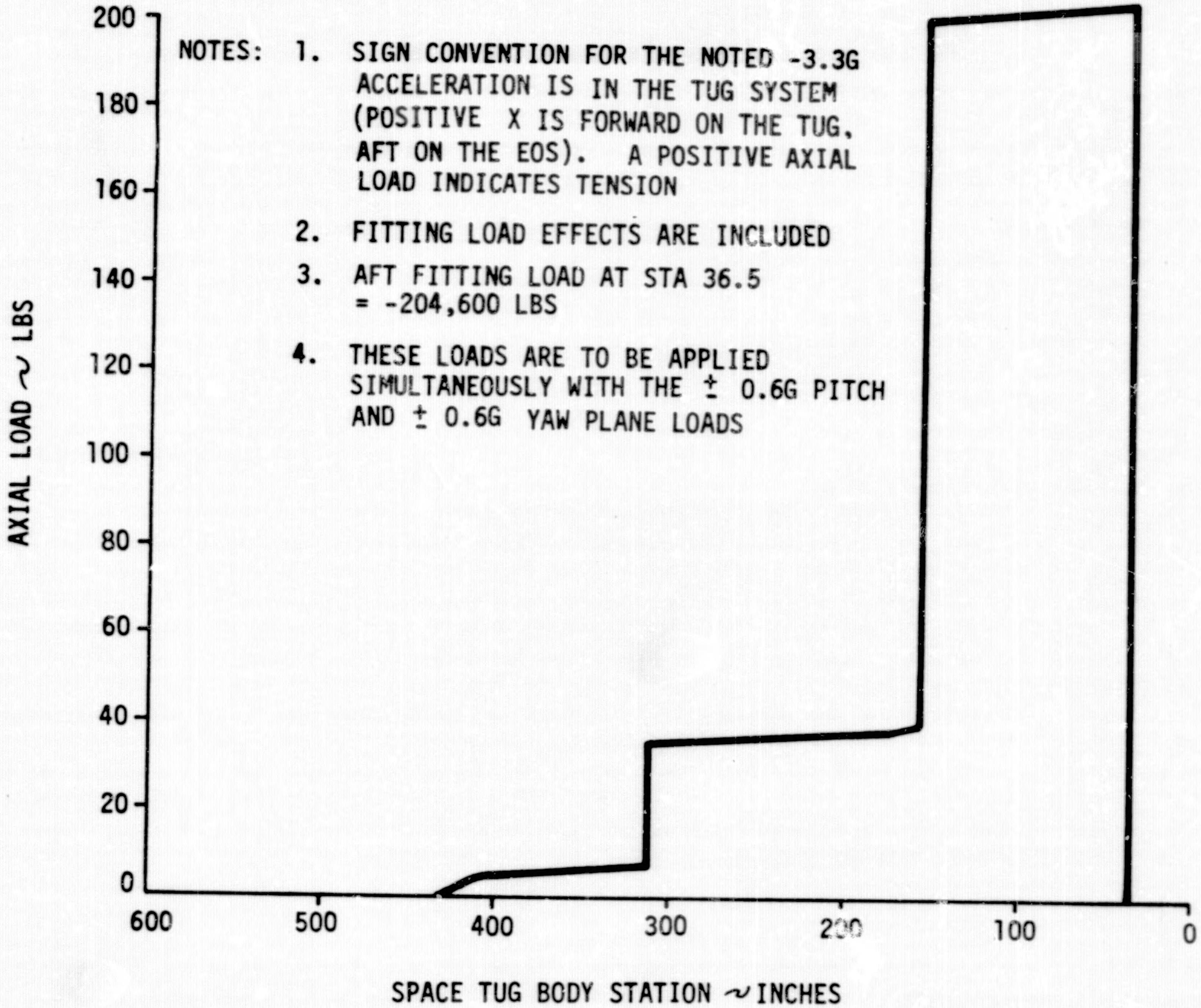


Figure 8.2-16 Space Tug (62K, No Payload) in Space Shuttle Cargo Bay -3.3G Limit Axial Body Loads

NOTES:

1. SIGN CONVENTION FOR THE NOTED 1.0G ACCELERATION IS IN THE TUG SYSTEM. FOR LOADS CORRESPONDING TO A 1.0G Y (YAW) ACCELERATION, THE SIGN OF THE BENDING MOMENT MUST BE MULTIPLIED BY -1.0.
2. FITTING LOAD EFFECTS ARE INCLUDED. THE CORRESPONDING FITTING LOAD AND TORQUE VALUES ARE PRESENTED IN TABLE 8.2-2
3. THESE LOADS ARE TO BE COMBINED WITH THE AXIAL LOADS OF THE -3.0G AND +0.2G CASES

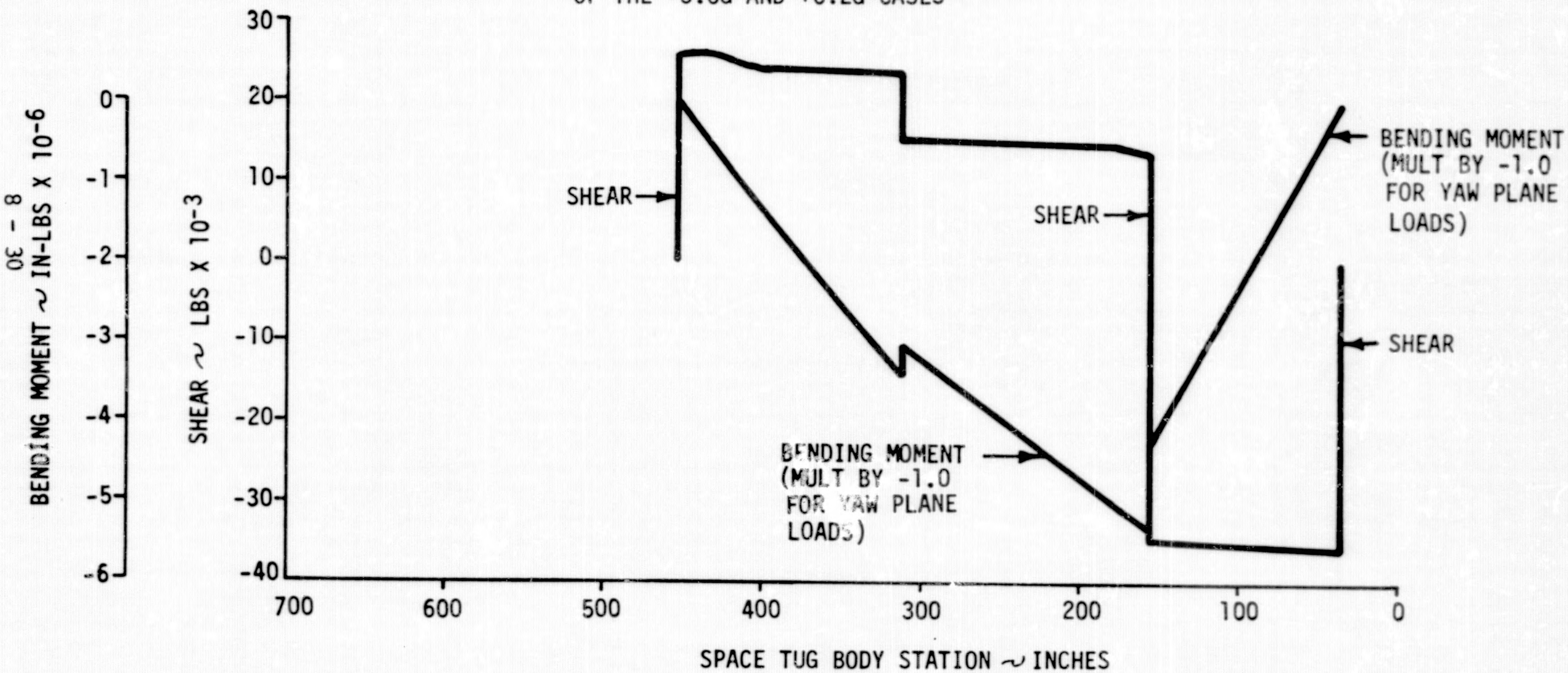


Figure 8.2-17 Space Tug (62K, No Payload) in Space Shuttle Cargo Bay 1.0G X-Z and X-Y Plane Limit Body Loads

NOTES: 1. SIGN CONVENTION FOR THE NOTED 0.6G ACCELERATION IS IN THE TUG SYSTEM. FOR LOADS CORRESPONDING TO A 0.6G (Y) YAW ACCELERATION, THE SIGN OF THE BENDING MOMENT MUST BE MULTIPLIED

2. FITTING LOAD EFFECTS ARE INCLUDED. THE CORRESPONDING FITTING LOAD AND TORQUE VALUES ARE PRESENTED IN TABLE 8.2-2

3. THESE LOADS ARE TO BE COMBINED WITH THE AXIAL LOADS OF THE -3.3G CASE

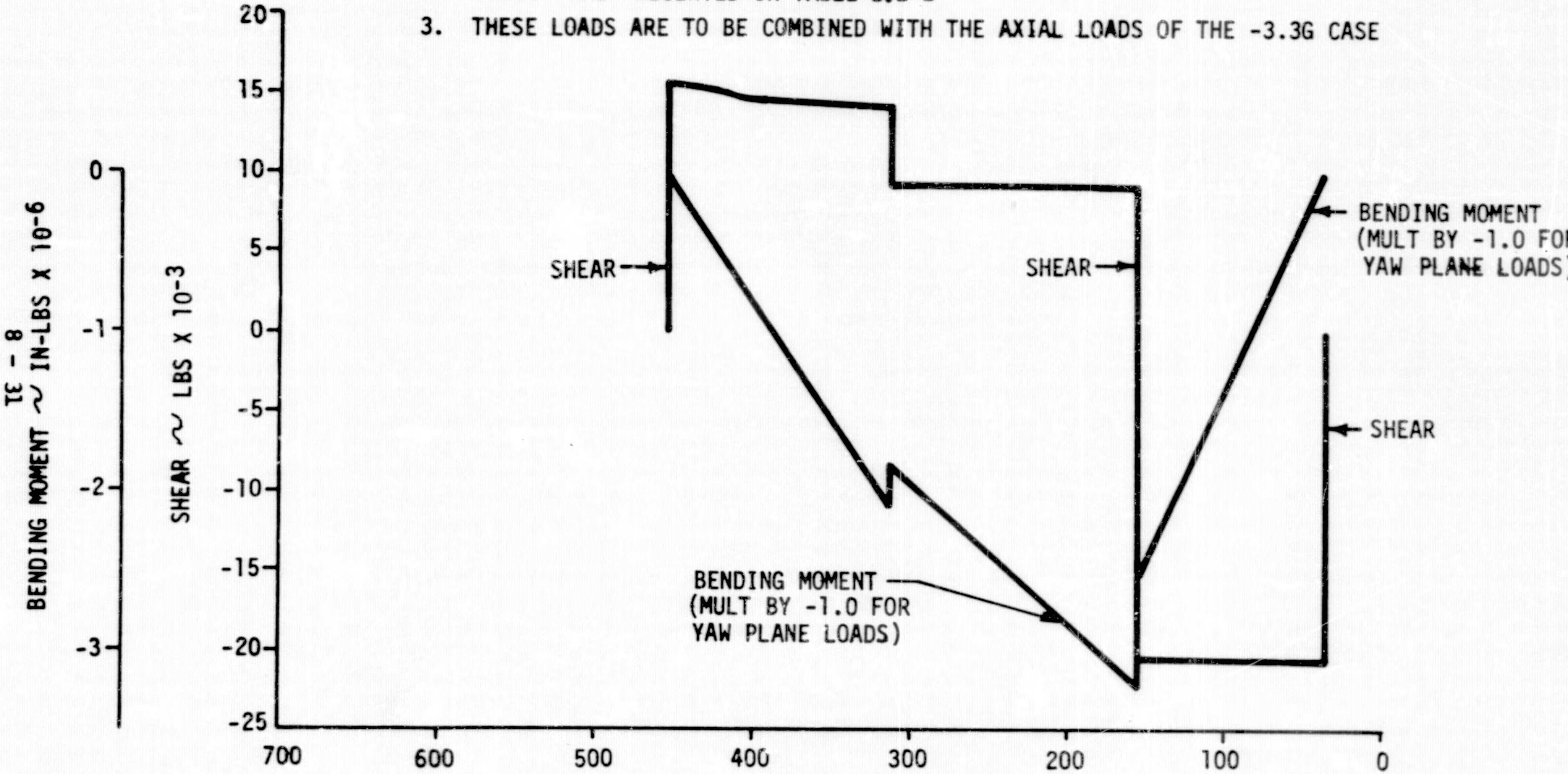


Figure 8.2-18 Space Tug (62K, No Payload) in Space Shuttle Cargo Bay 0.6G X-Z and X-Y Plane Limit Body Loads

TABLE 8.2-3

TUG POINT DESIGN TEMPERATURE ENVIRONMENT

Condition	TEMPERATURES (°F)	
	Minimum	Maximum
Pre-launch	-100	+120
Launch	-100	+120
On-Orbit (Orbiter bay)	-100	+200
On-Orbit (Space)	(-50)*	(-50)*
Entry and post landing	-100	+200
(*Nominal value to be used while awaiting mission definiton.)		

longerons and face sheet reinforcement to react the concentrated loads. The thrust structure is a conical arrangement of frame and skin-stabilized tubular struts attached to the LOX tank aft bulkhead. In addition to being the support structure for the Tug main engine, this structure provides attachment for engine feed lines and equipment, and micro-meteoroid protection for the aft end of the Tug. Construction is of advanced composites, primarily graphite-epoxy with Boron-epoxy used for the longitudinal fibers of the tubular struts in order to reduce heat leak into the LOX tank.

8.3.1 Shell Structure

General Description

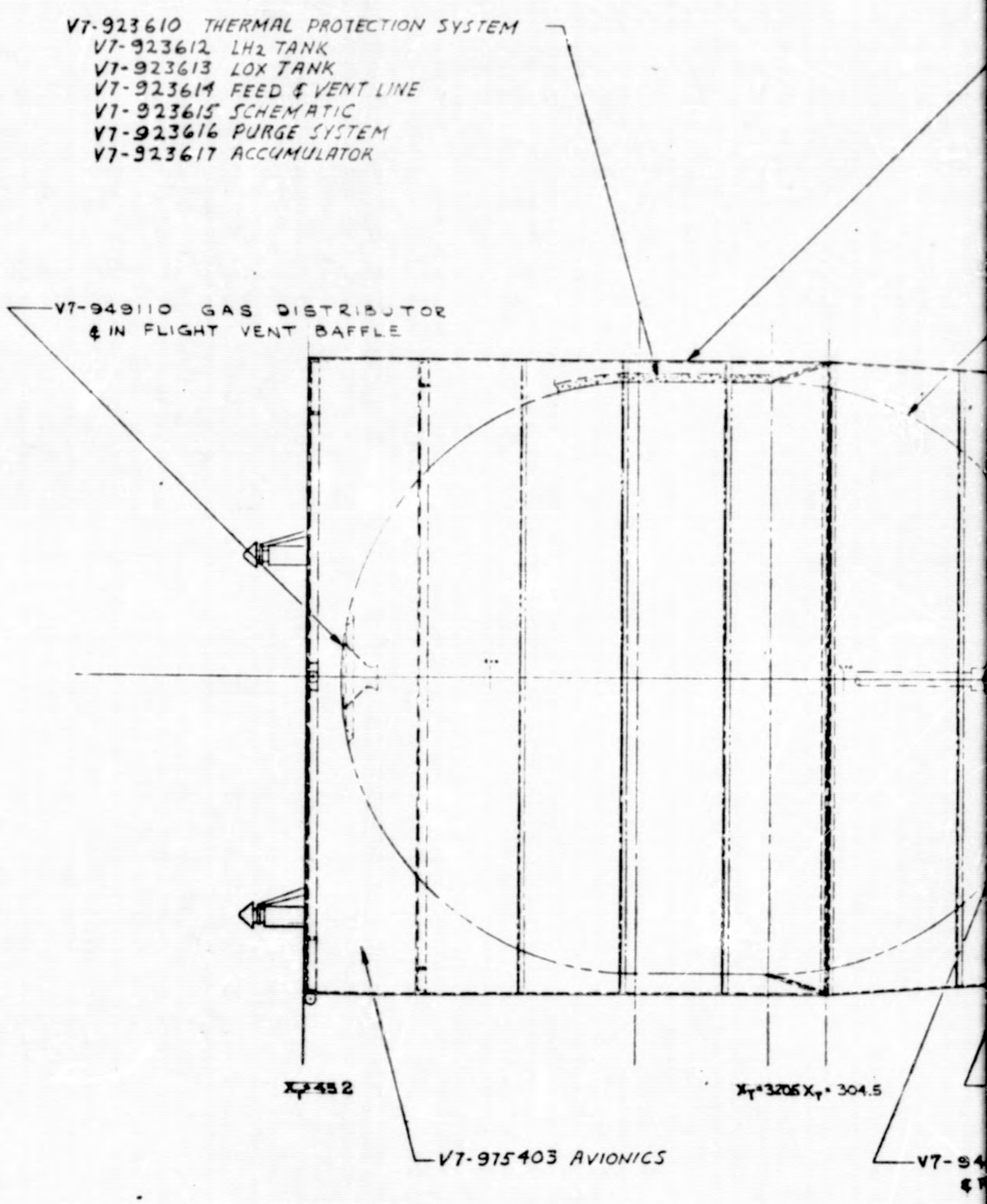
The Space Tug shell design is defined by six drawings. Three of these, Figures 8.3-1, 8.3-2, and 1.5-1 of this volume, define the orientation and general arrangement of the shell and vehicle system. The remaining drawings provide detail designs of specific components. Figures 8.3-3, 8.3-4 and 8.3.5 show the forward skirt and payload docking, the intertank structure, and the aft skirt and engine thrust structure, respectively.

The non-integral structural shell of the Space Tug is 322 inches long. This shell is divided into three sections - a cylindrical forward skirt of 90-inch radius, a cylindrical aft skirt of 81-inch radius, and a cone frustrum, intertank shell connecting the two cylindrical sections. All sections of the shell are of honeycomb sandwich construction. Internal frames are used for shell stability, distribution of concentrated loads, and support of system hardware. Major frames are located at the ends of the shell (Stations 126 and 452) and at the interface of shell sections. These interface frames are at field joints. They provide support points for tank attachment struts as well as a splice for adjacent shell sections. Stability frames are used to reduce the critical length of the shell on the basis of general stability and weight minimization. Each shell section is divided into four equal arc sectors

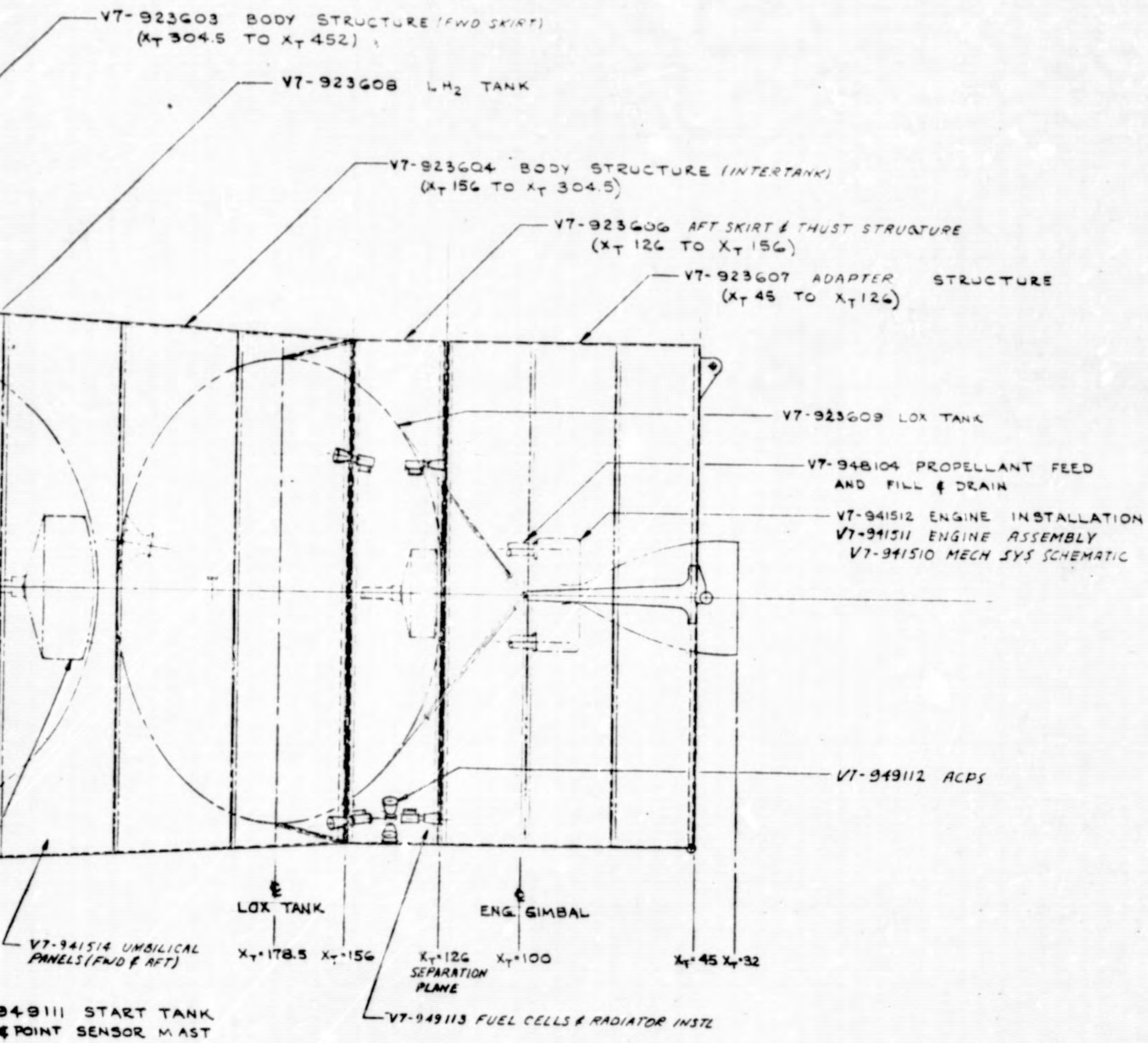
TABLE 8.2-4

TUG INTERNAL ACOUSTIC DESIGN CRITERIA
SOUND PRESSURE LEVEL (db) REF. 10^{-5} N/M^2

1/3 Octave Center Band Freq. (Hz)	Lift-Off	Boundary Layer
5	124	124.5
6.3	127	125.0
8	128	126.0
10	129	126.5
12.5	131	127.0
16	132	128.0
20	134	128.5
25	135	129.0
31.5	137	130.0
40	138	130.5
50	139	131.0
63	140	132.0
80	141	132.5
100	143	133.0
125	144	134.0
160	145	134.5
200	145	135.5
250	145	136.0
315	144	136.5
400	143	137.0
500	142	137.5
630	141	138.0
800	140	138.5
1K	139	138.0
1.25K	138	137.0
1.6K	137	136.5
2K	135	135.5
2.5K	134	134.5
3.15K	133	134.0
4K	132	133.0
5K	131	132.0
6.3K	130	131.0
8K	129	130.0
10K	128	129.0
	OASPL155db	OASPL149db


V7-923610 THERMAL PROTECTION SYSTEM
V7-923612 LH2 TANK
V7-923613 LOX TANK
V7-923614 FEED & VENT LINE
V7-923615 SCHEMATIC
V7-923616 PURGE SYSTEM
V7-923617 ACCUMULATOR
V7-949110 GAS DISTRIBUTOR
& IN FLIGHT VENT BAFFLE
V7-975403 AVIONICS
V7-94

F-1

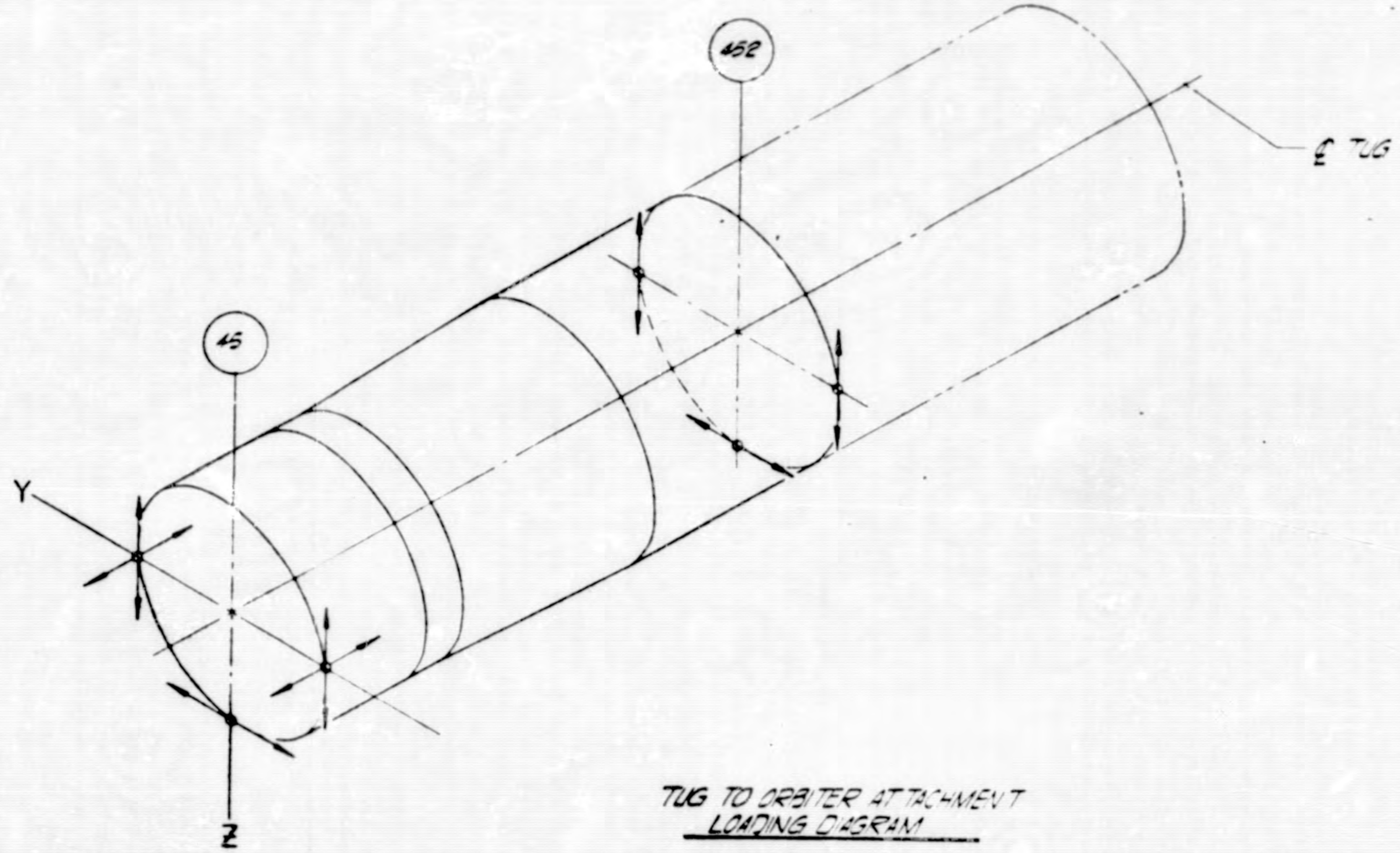


INFORMATION ONLY

Figure 8.3-1 Tug Diagram

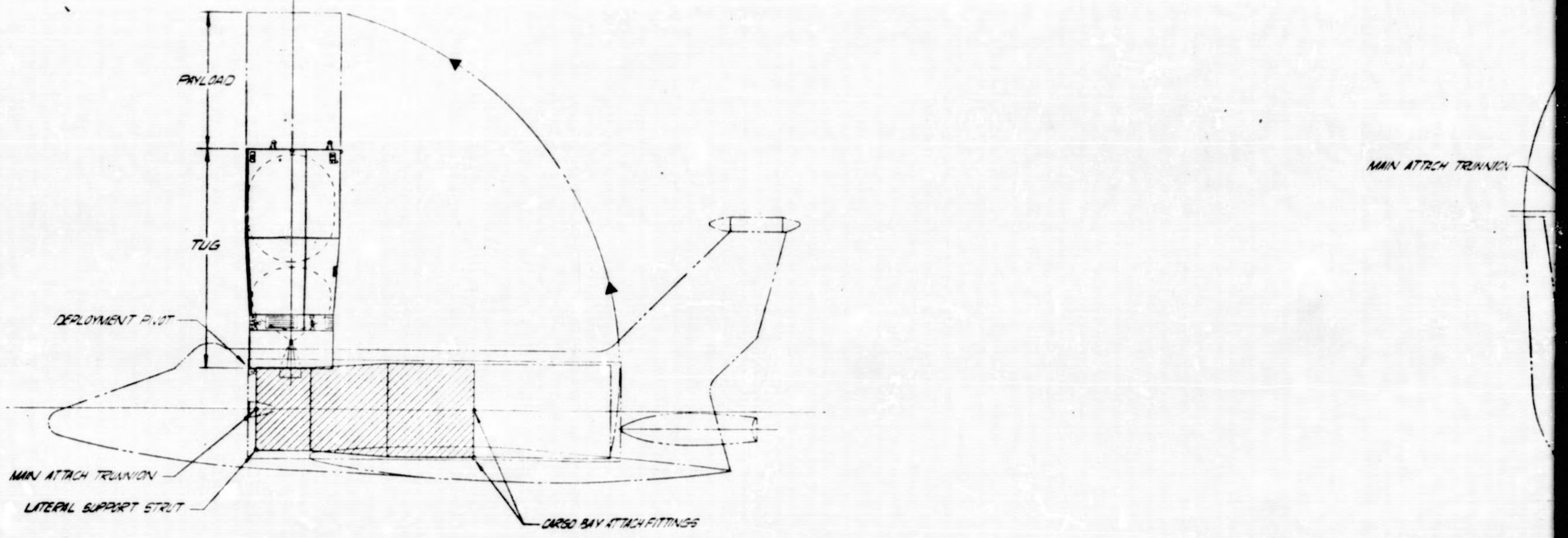
F-2

1/20	SWENSON
DATE	16 DEC 71
SPACE DIVISION	NORTH AMERICAN ROCKWELL CORPORATION, 12214 LAKEWOOD BOULEVARD, DOWNEY CALIFORNIA
TUG DIAGRAM	V7-923601



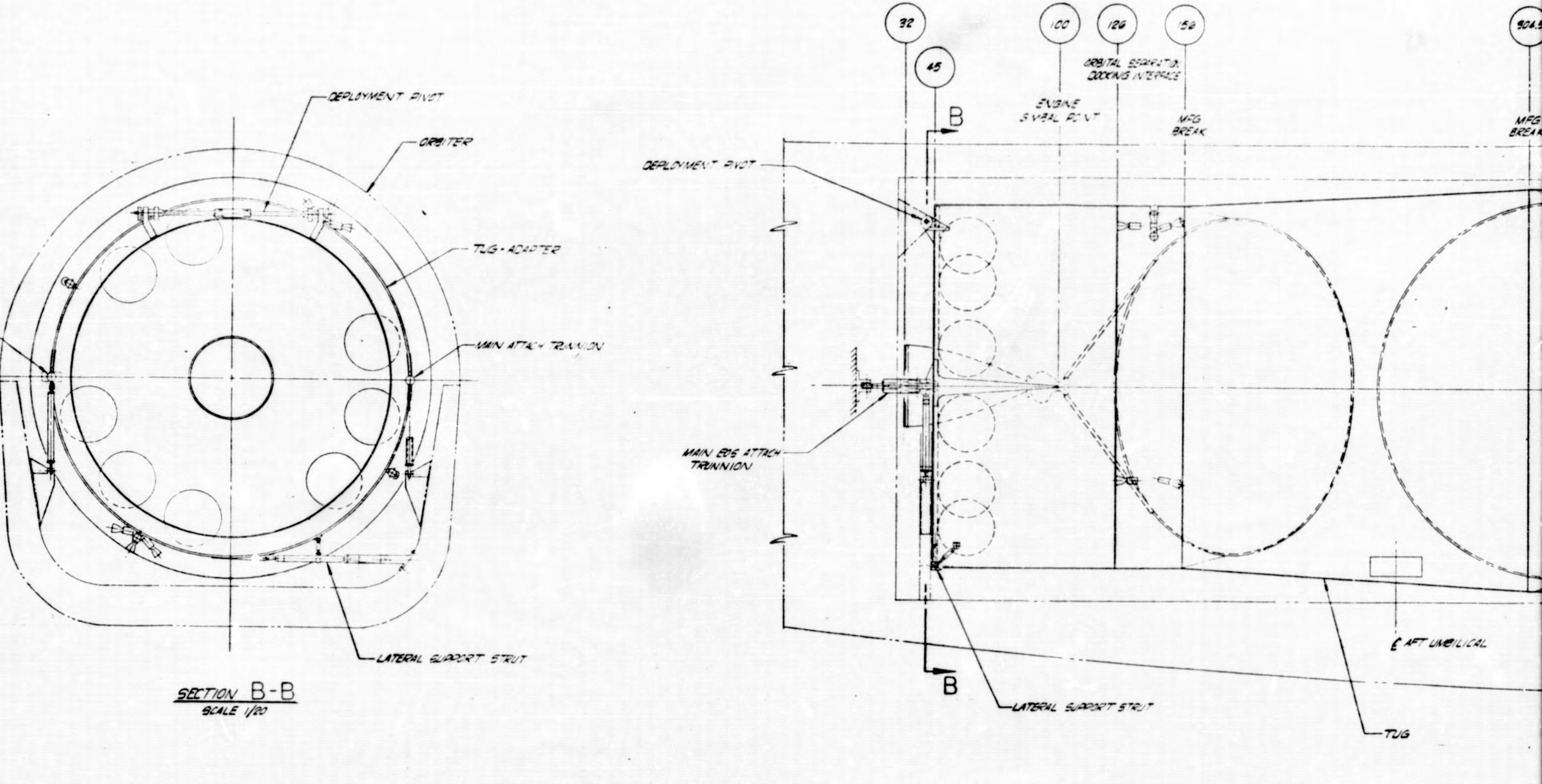
TUG TO ORBITER ATTACHMENT
LOADING DIAGRAM

F-1

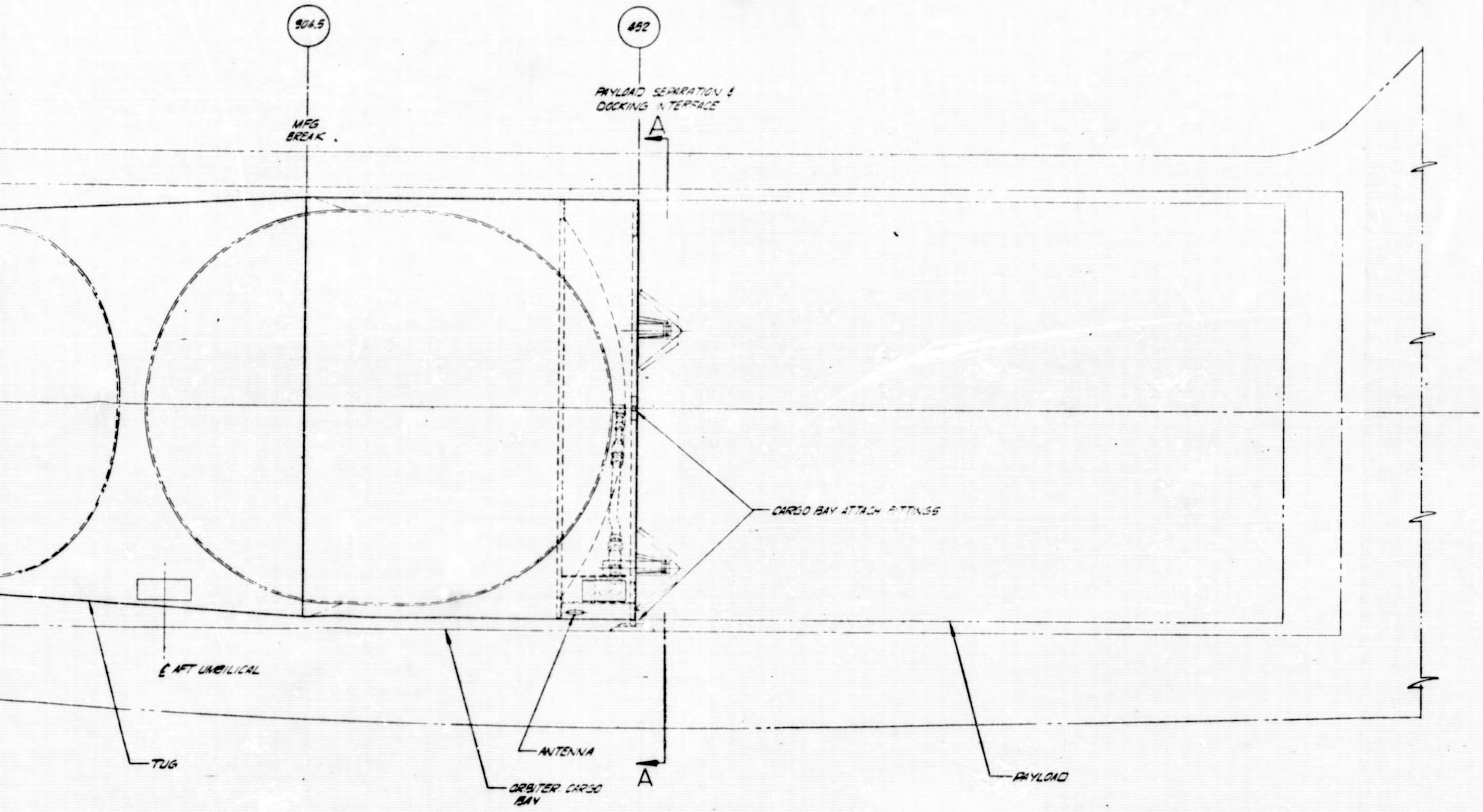


TUG/ORBITER ORIENTATION
SCALE 1/100

F-2


DEPLOYMENT PIVOT
ORBITER
TUG - ADAPTER
MAIN ATTACH TRUNNION
LATERAL SUPPORT STRUT
SECTION B-B
SCALE 1/20
32
45
100
126
156
ORBITAL SEPARATION DOCKING INTERFACE
ENGINE GIMBAL POINT
MFG BREAK
MFG BREAK
B
B
DEPLOYMENT PIVOT
MAIN EOS ATTACH TRUNNION
LATERAL SUPPORT STRUT
℄ AFT UMBILICAL
TUG

F-3



904.5
952
PAYLOAD SEPARATION & DOCKING INTERFACE
MFG BREAK
A
CARGO BAY ATTACH FITTINGS
℄ AFT UMBILICAL
TUG
ANTENNA
ORBITER CARGO BAY
A
PAYLOAD

F-4

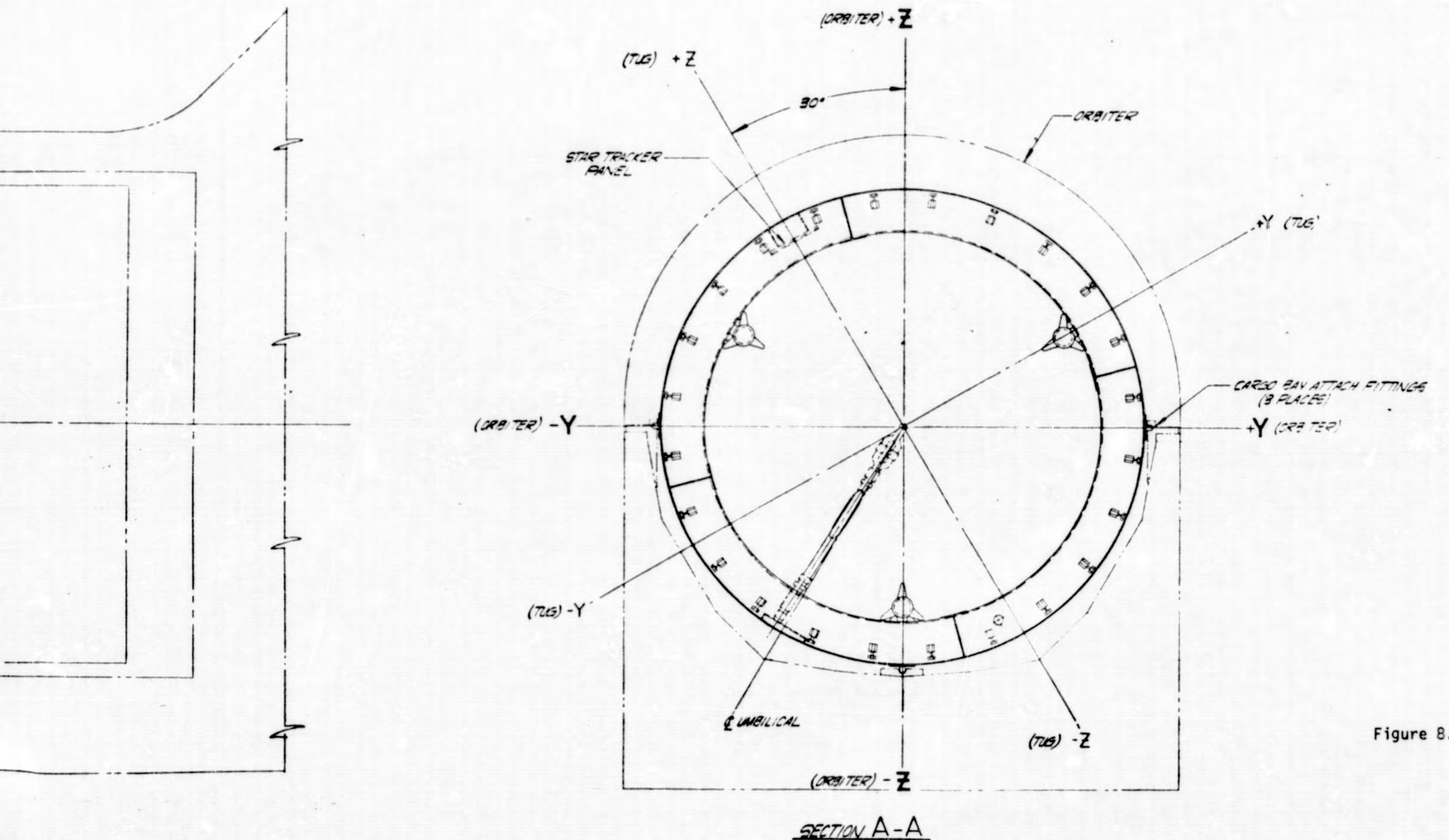
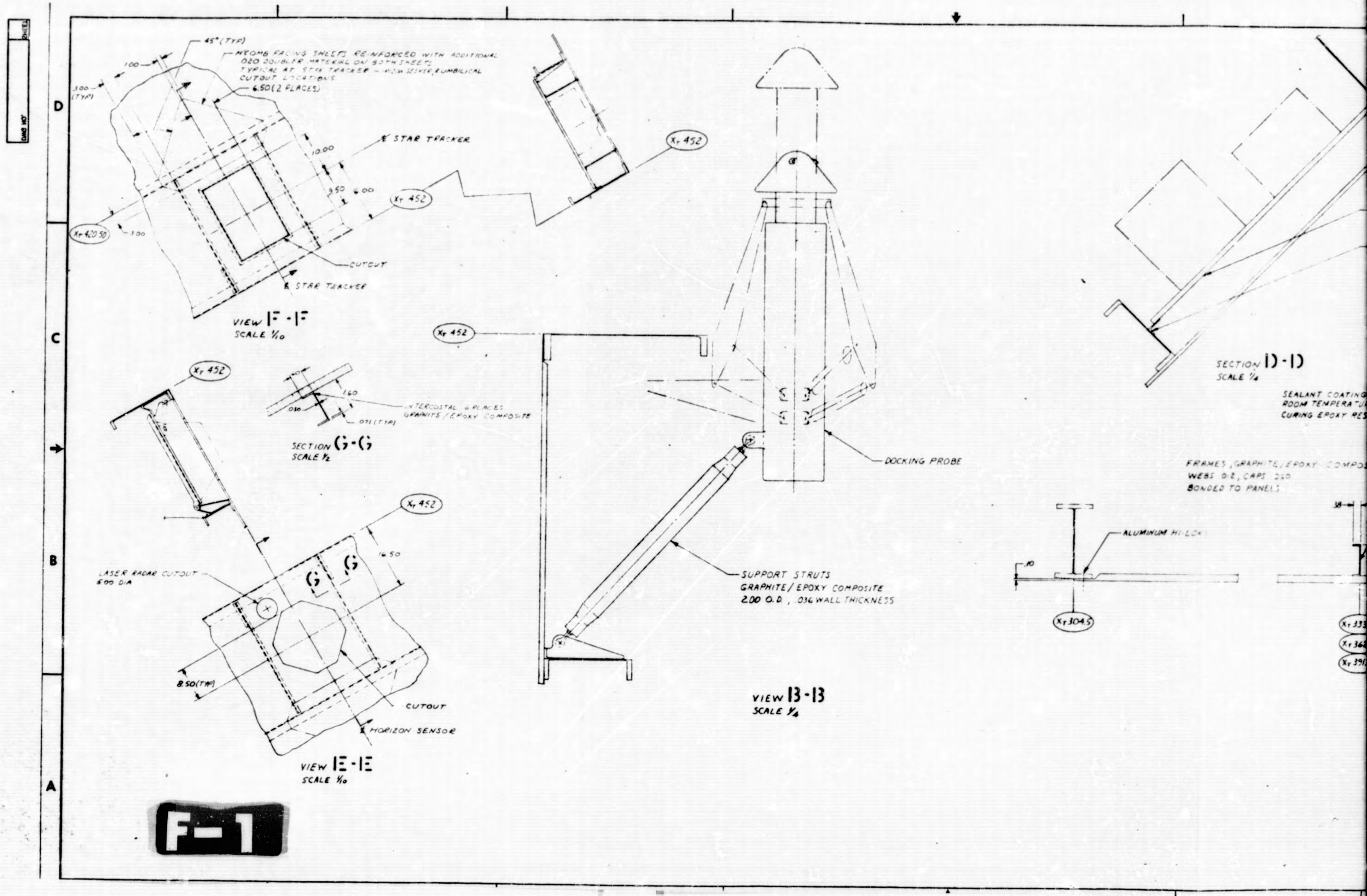


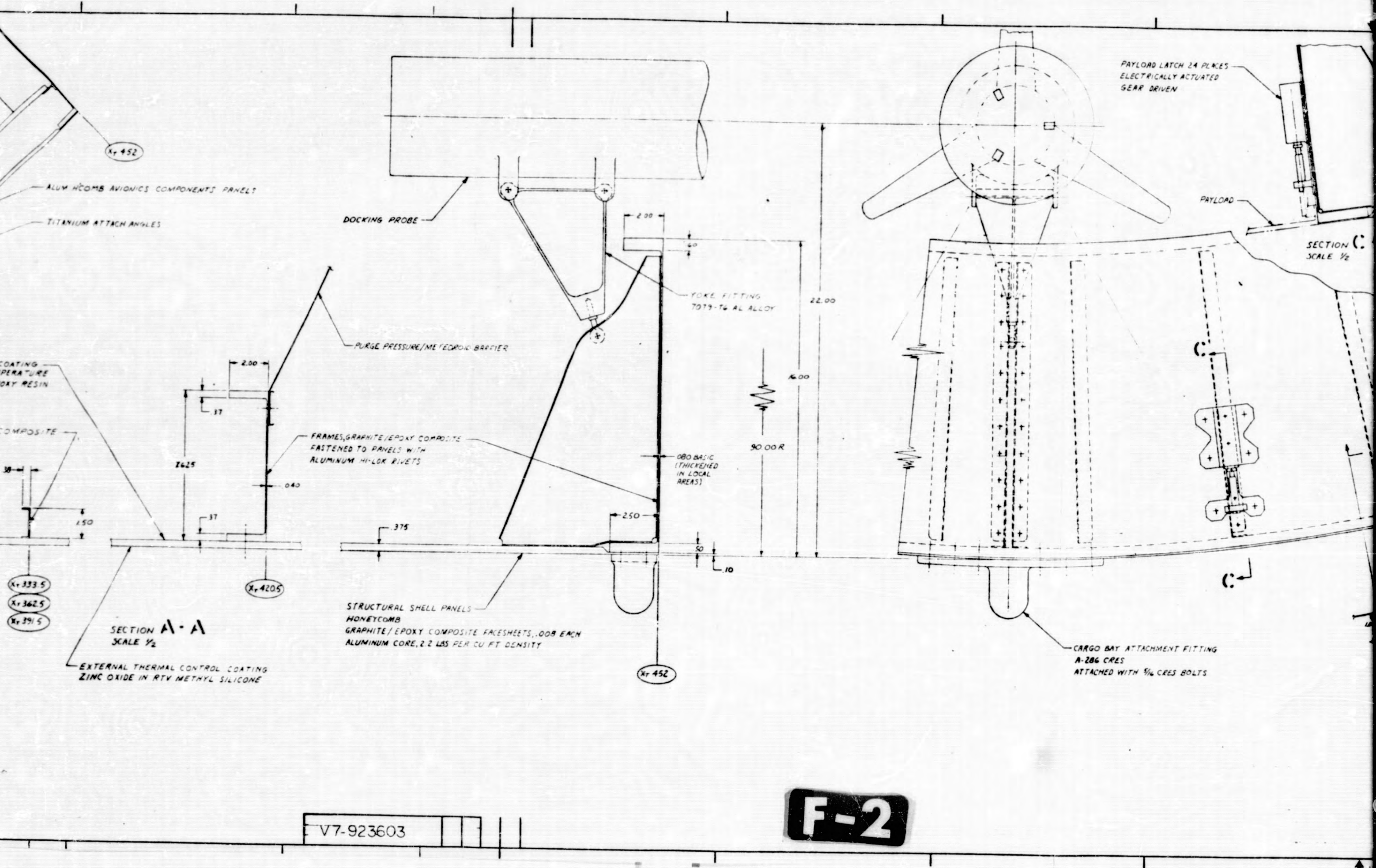
Figure 8.3-2 Orientation - Tug to Shuttle Cargo Bay

F-5

1/20	NORTH AMERICAN ROCKWELL CORPORATION	
ORIENTATION - TUG TO SHUTTLE CARGO BAY		V7-923605

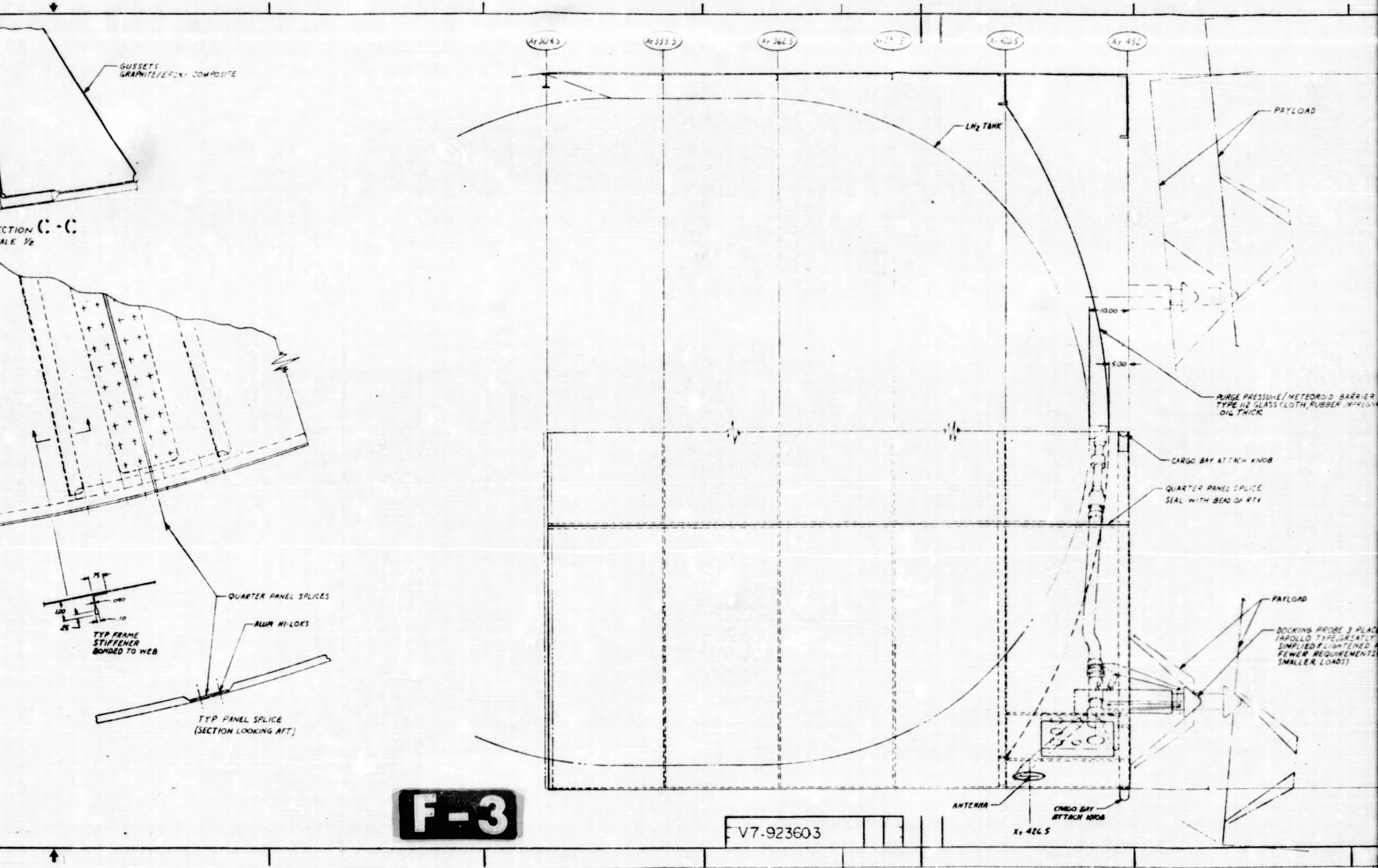

45° (TYP)
.100
.300 (TYP)
6.50 (2 PLACES)
℄ STAR TRACKER
10.00
3.50
6.00
XT 452
XT 420.50
.300
CUTOUT
℄ STAR TRACKER
VIEW F-F
SCALE 1/10
XT 452
XT 452
1.60
.030
.071 (TYP)
INTERCOSTAL 4 PLACES
GRAPHITE / EPOXY COMPOSITE
SECTION G-G
SCALE 1/2
XT 452
16.50
LASER RADAR CUTOUT
6.00 DIA
G
G
8.50 (TYP)
CUTOUT
℄ HORIZON SENSOR
VIEW E-E
SCALE 1/10
XT 452
DOCKING PROBE
SUPPORT STRUTS
GRAPHITE / EPOXY COMPOSITE
2.00 O.D., .036 WALL THICKNESS
VIEW B-B
SCALE 1/4
SECTION D-D
SCALE 1/4
SEALANT COATING
ALUMINUM HI-LOK
.10
XT 304.5

F-1



XT 452
ALUM H'COMB AVIONICS COMPONENTS PANELS
TITANIUM ATTACH ANGLES
DOCKING PROBE
2.00
.40
YOKE FITTING
7075-T6 AL ALLOY
22.00
16.00
PURGE PRESSURE/METEOROID BARRIER
2.00
.37
2.625
.040
.38
.150
.37
FRAMES, GRAPHITE/EPOXY COMPOSITE
FASTENED TO PANELS WITH
ALUMINUM HI-LOK RIVETS
.080 BASIC
(THICKENED
IN LOCAL
AREAS)
90.00 R
2.50
.375
.50
.10
XT 333.5
XT 362.5
XT 391.5
XT 420.5
SECTION A-A
SCALE 1/2
STRUCTURAL SHELL PANELS
HONEYCOMB
GRAPHITE/EPOXY COMPOSITE FACESHEETS, .008 EACH
ALUMINUM CORE, 2.2 LBS PER CU FT DENSITY
XT 452
EXTERNAL THERMAL CONTROL COATING
ZINC OXIDE IN RTV METHYL SILICONE
PAYLOAD LATCH 24 PLACES
ELECTRICALLY ACTUATED
GEAR DRIVEN
PAYLOAD
SECTION C
SCALE 1/2
C
C
CARGO BAY ATTACHMENT FITTING
A-286 CRES
ATTACHED WITH 5/16 CRES BOLTS
V7-923603

F-2



GUSSETS GRAPHITE/EPOXY COMPOSITE
SECTION C-C
SCALE 1/2
QUARTER PANEL SPLICES
ALUM HI-LOKS
TYP FRAME STIFFENER BONDED TO WEB
TYP PANEL SPLICE (SECTION LOOKING AFT)
Xr 304.5
Xr 333.5
Xr 362.5
Xr 420.5
Xr 452
LH2 TANK
PAYLOAD
10.00
5.00
PURGE PRESSURE/METEOROID BARRIER TYPE 112 GLASS CLOTH, RUBBER IMPREGN
.016 THICK
CARGO BAY ATTACH KNOB
QUARTER PANEL SPLICE SEAL WITH BEAD OF RTV
PAYLOAD
DOCKING PROBE 3 PLAC (APOLLO TYPE, GREATLY SIMPLIFIED & LIGHTENED FEWER REQUIREMENTS SMALLER LOADS)
ANTENNA
CARGO BAY ATTACH KNOB
Xr 426.5
F-3
V7-923603

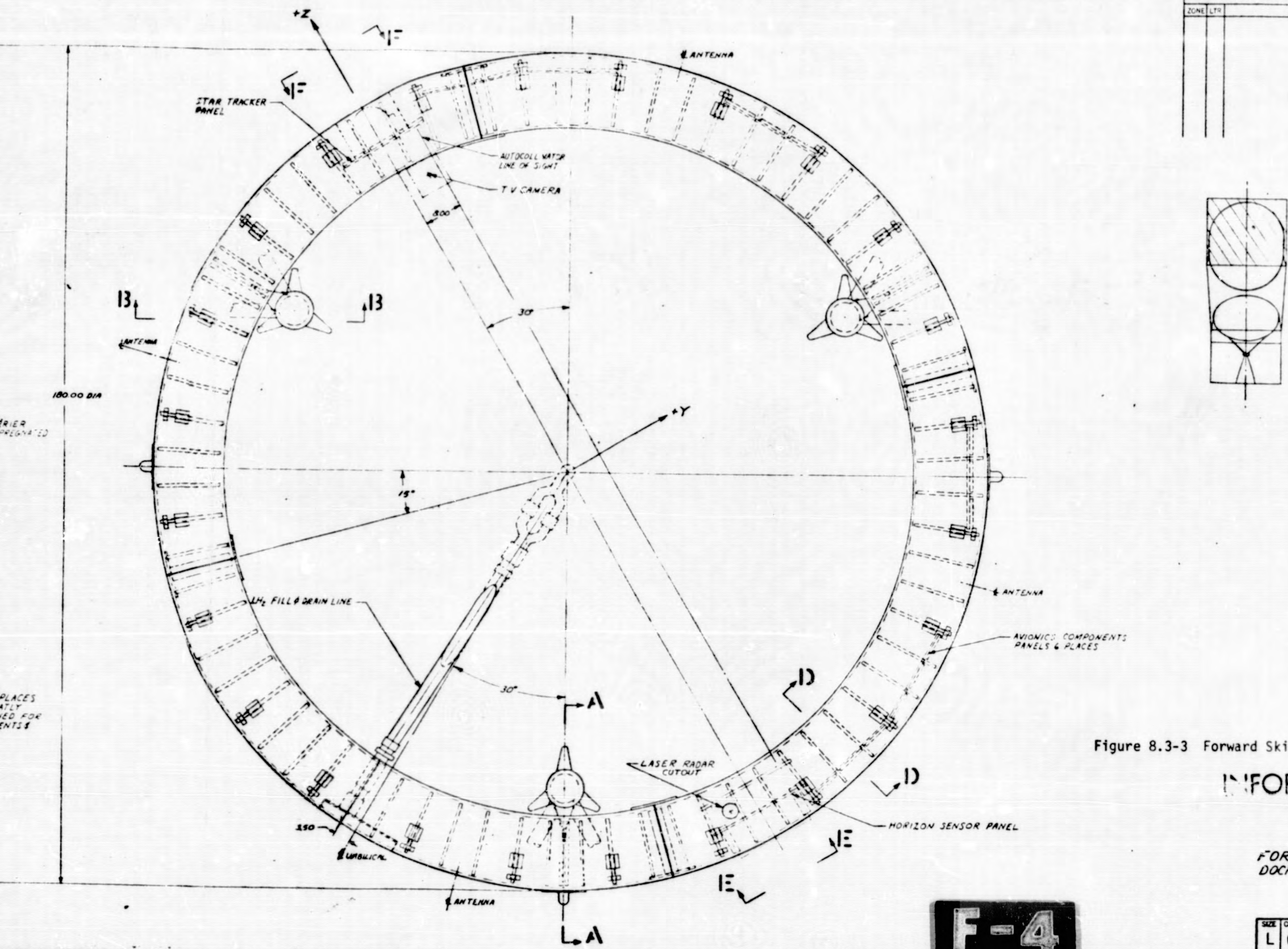


Figure 8.3-3 Forward Skirt & Forward Docking - Tug

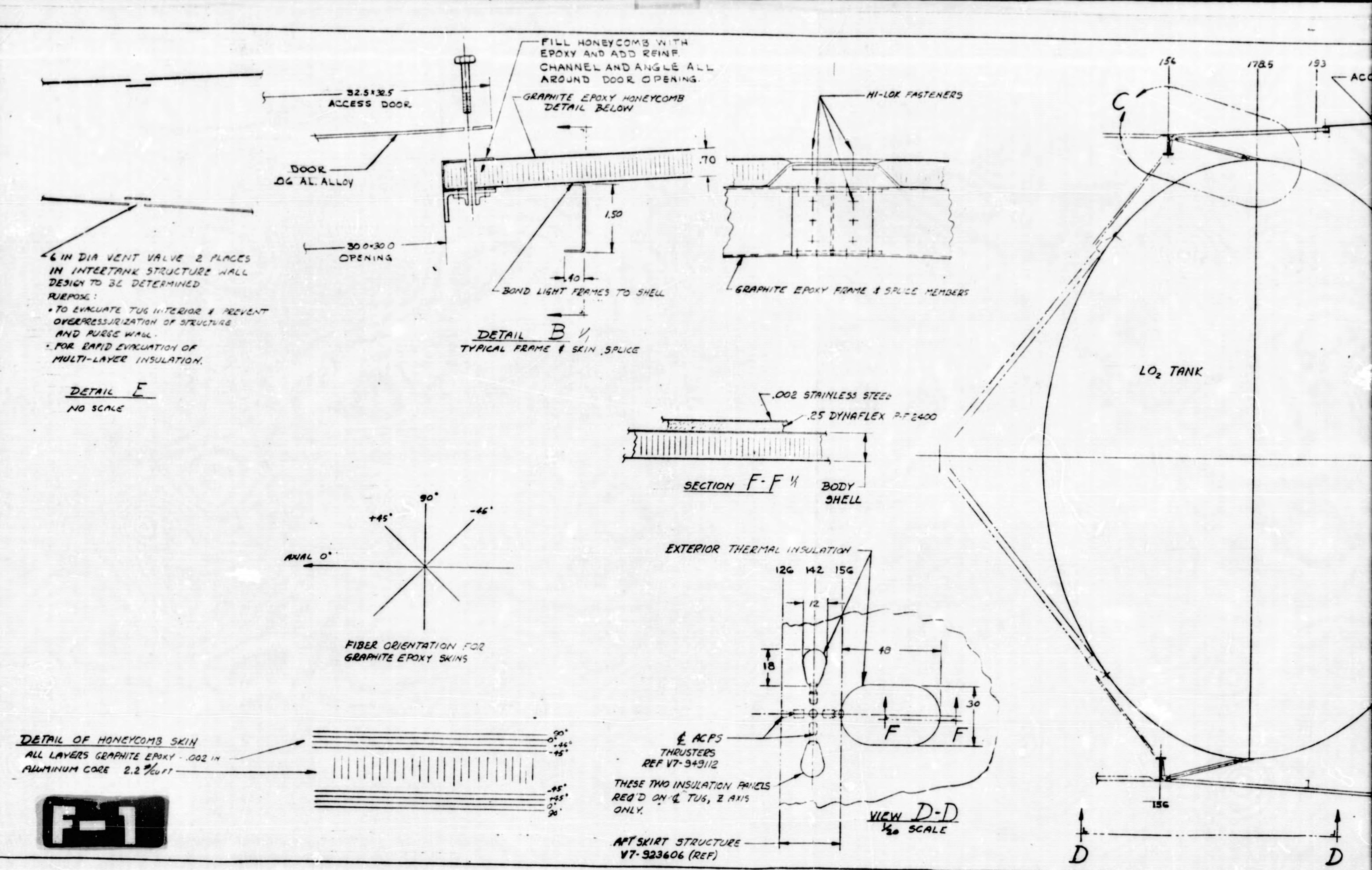
INFORMATION ONLY

FORWARD SKIRT & FORWARD DOCKING - TUG

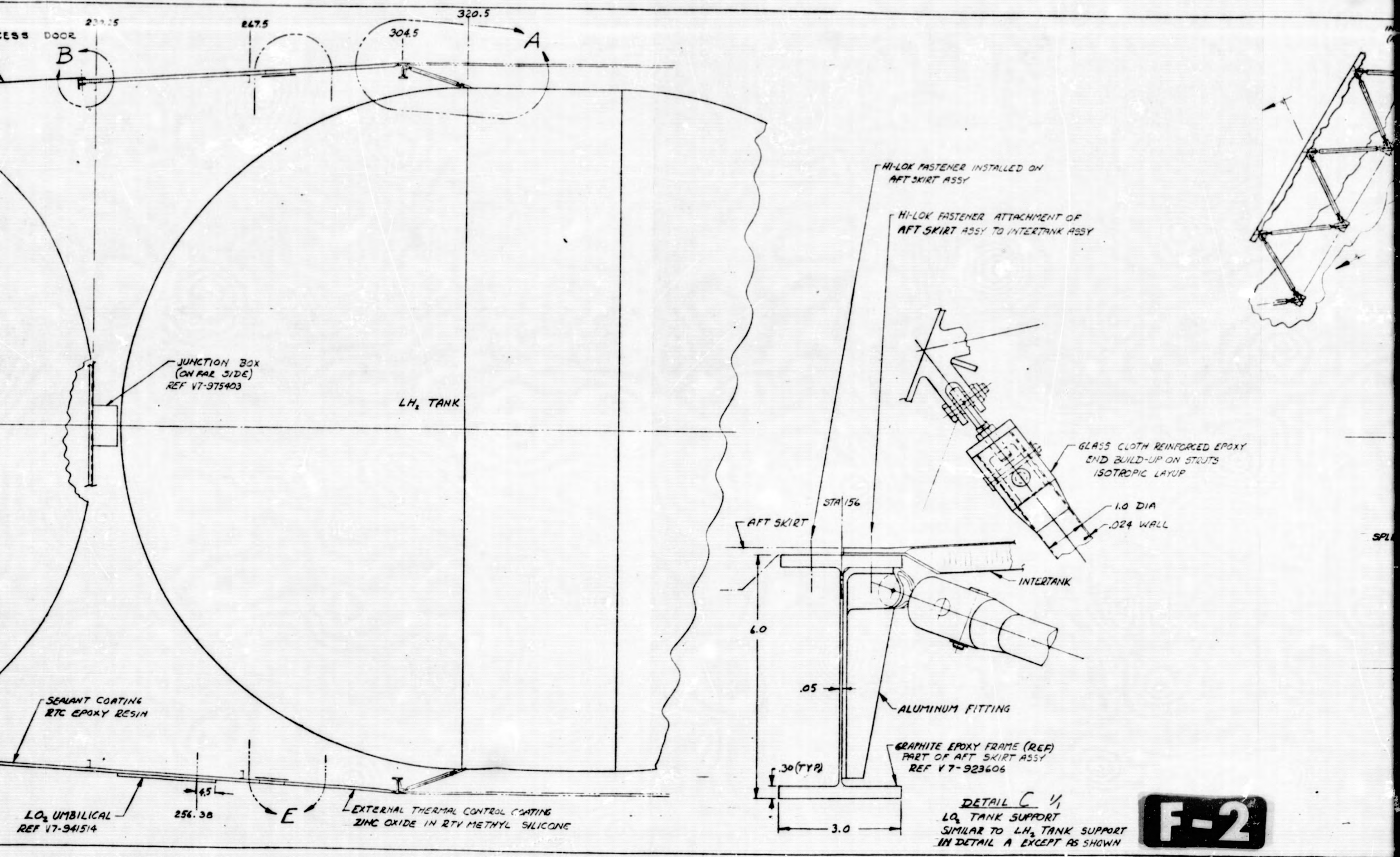
SIZE	CODE IDENT NO.	DRAWING NO.
L	03953	V7-923603

V7-923603

F-4

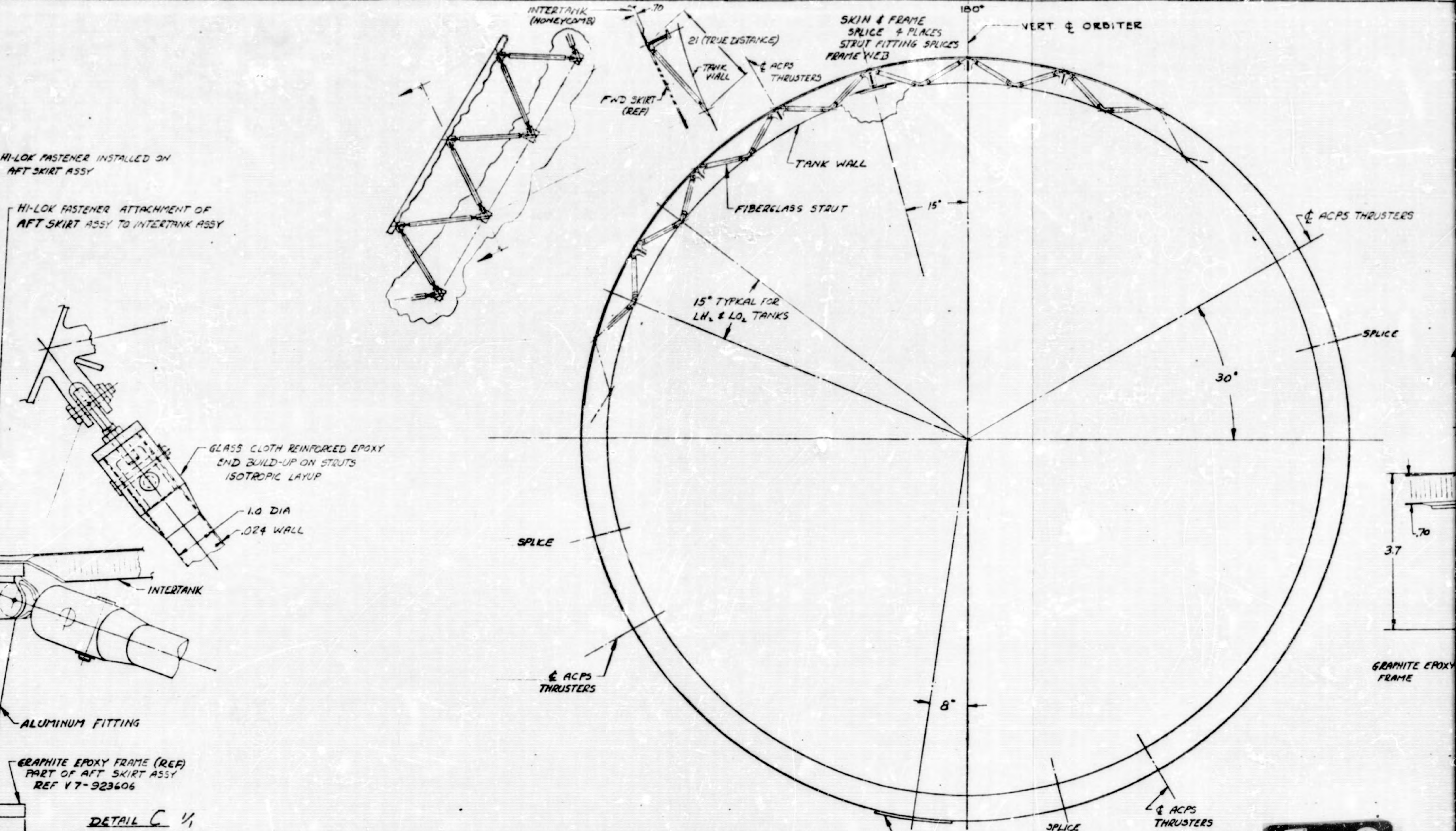


FILL HONEYCOMB WITH EPOXY AND ADD REINF. CHANNEL AND ANGLE ALL AROUND DOOR OPENING.
32.5×32.5 ACCESS DOOR
GRAPHITE EPOXY HONEYCOMB DETAIL BELOW
DOOR .06 AL. ALLOY
.70
1.50
30.0×30.0 OPENING
.40
BOND LIGHT FRAMES TO SHELL
DETAIL B 1/1
TYPICAL FRAME & SKIN SPLICE
HI-LOK FASTENERS
GRAPHITE EPOXY FRAME & SPLICE MEMBERS
6 IN DIA VENT VALVE 2 PLACES IN INTERTANK STRUCTURE WALL
DESIGN TO BE DETERMINED
PURPOSE:
• TO EVACUATE TUG INTERIOR & PREVENT OVERPRESSURIZATION OF STRUCTURE AND PURGE WALL.
• FOR RAPID EVACUATION OF MULTI-LAYER INSULATION.
DETAIL E
NO SCALE
.002 STAINLESS STEEL
.25 DYNAFLEX P-F 2400
SECTION F-F 1/1
BODY SHELL
90°
+45°
-45°
AXIAL 0°
FIBER ORIENTATION FOR GRAPHITE EPOXY SKINS
EXTERIOR THERMAL INSULATION
126 142 156
12
48
18
30
F F
℄ ACPS THRUSTERS REF V7-949112
THESE TWO INSULATION PANELS REQ'D ON ℄ TUG, Z AXIS ONLY.
VIEW D-D
1/20 SCALE
AFT SKIRT STRUCTURE V7-923606 (REF)
DETAIL OF HONEYCOMB SKIN
ALL LAYERS GRAPHITE EPOXY - .002 IN
ALUMINUM CORE 2.2 #/CU FT
90°
0°
+45°
-45°
156
178.5
193
C
LO$_2$ TANK
156
D
D
F-1



DETAIL C 1/1
LO_2 TANK SUPPORT
SIMILAR TO LH_2 TANK SUPPORT IN DETAIL A EXCEPT AS SHOWN

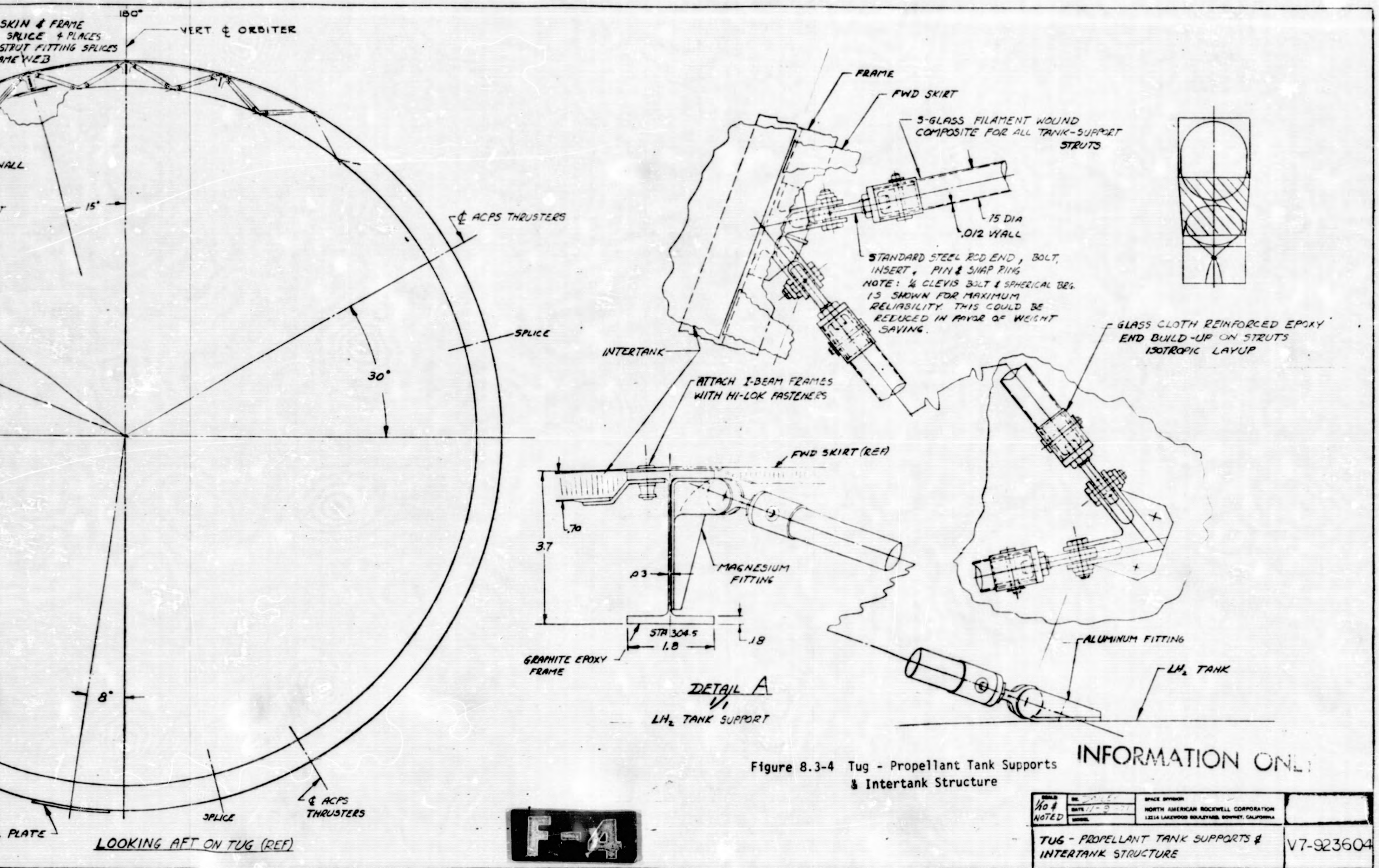
F-2



DETAIL C 1/1
LO_2 TANK SUPPORT
SIMILAR TO LH_2 TANK SUPPORT IN DETAIL A EXCEPT AS SHOWN

LOOKING AFT ON TUG (REF)

F-3

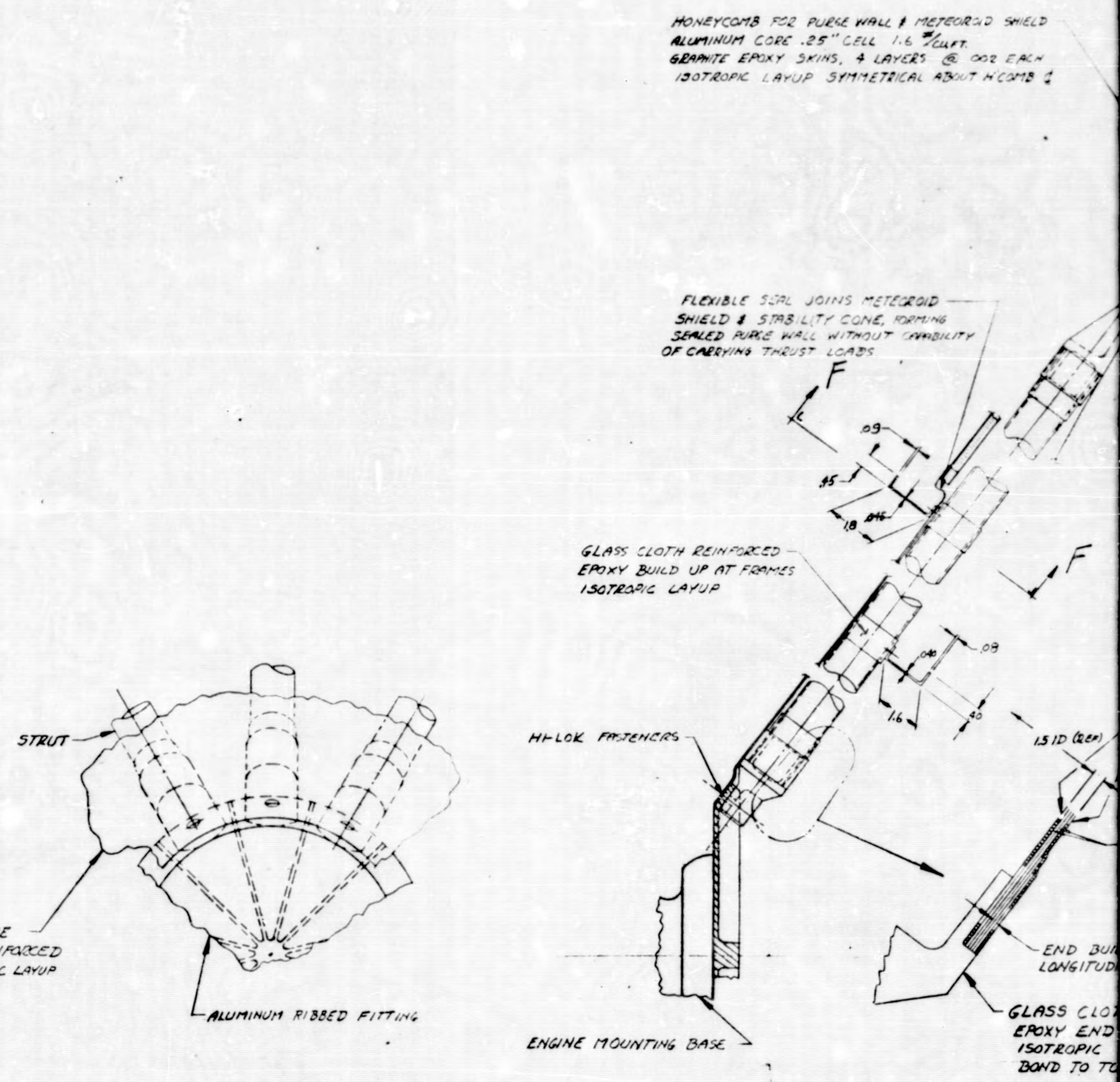


INFORMATION ONLY

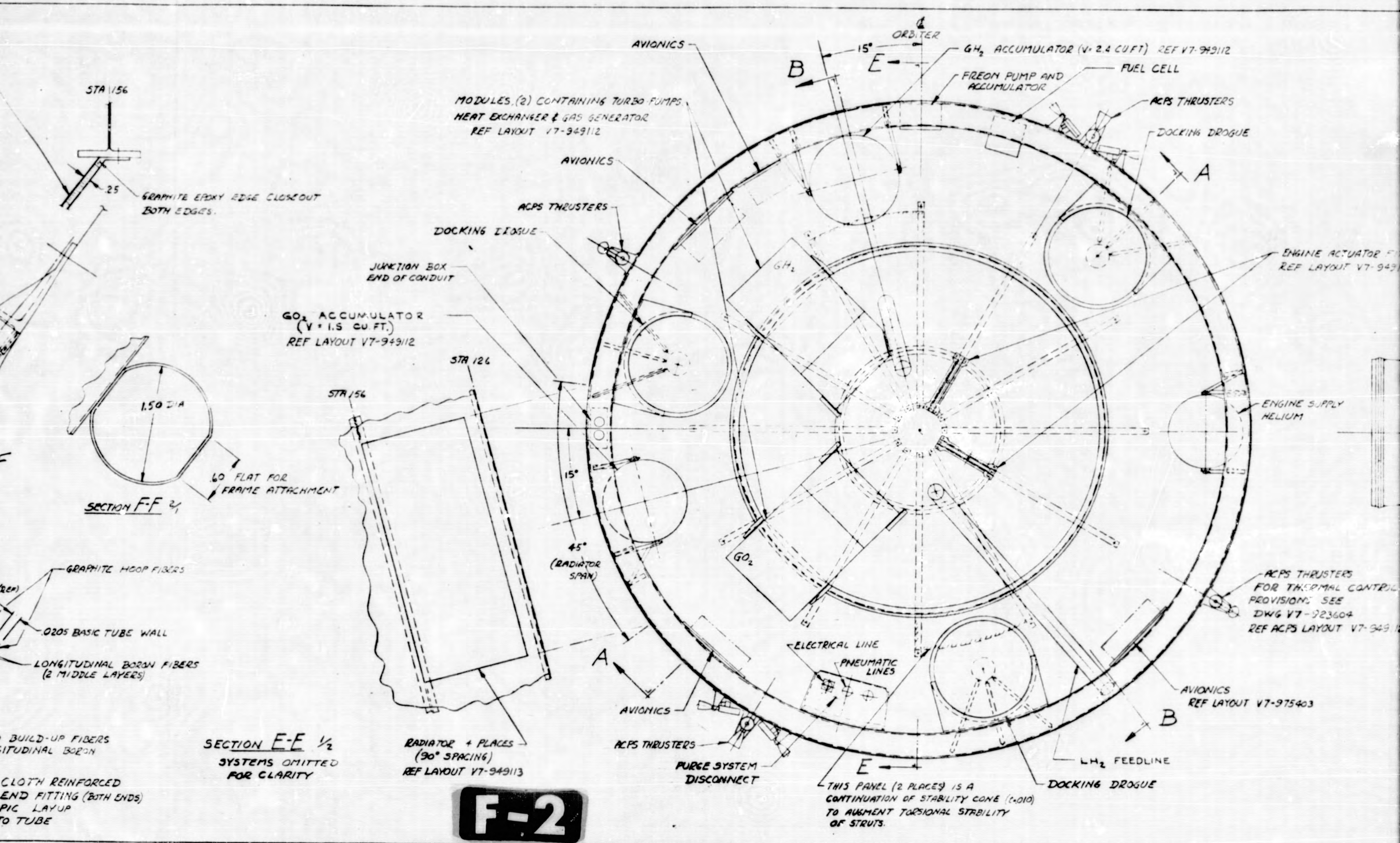
Figure 8.3-4 Tug - Propellant Tank Supports & Intertank Structure

1/10 & NOTED		SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
TUG - PROPELLANT TANK SUPPORTS & INTERTANK STRUCTURE		V7-923604

F-4

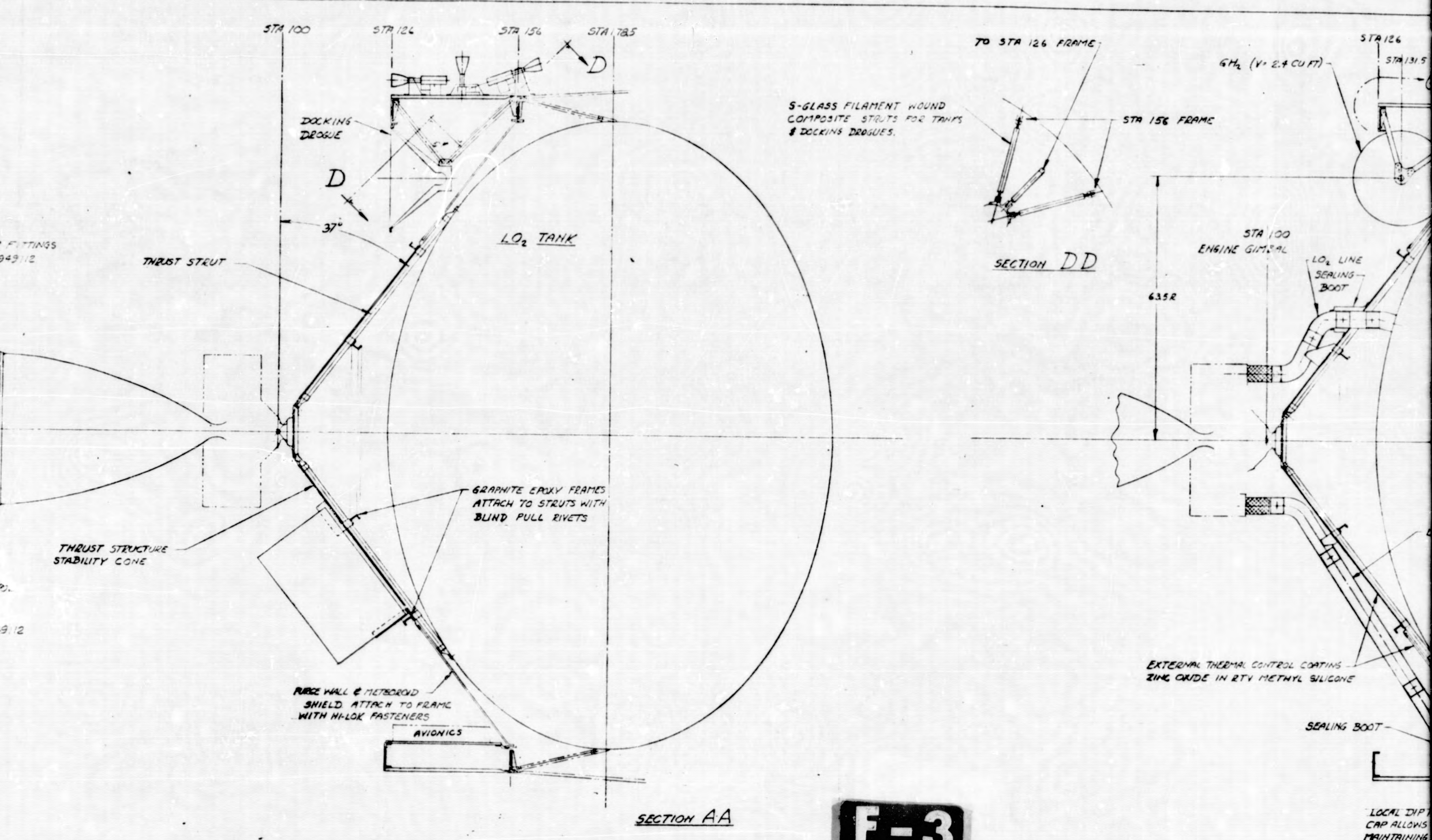

HONEYCOMB FOR PURGE WALL & METEOROID SHIELD
ALUMINUM CORE .25" CELL 1.6 #/CU.FT.
GRAPHITE EPOXY SKINS, 4 LAYERS @ .002 EACH
ISOTROPIC LAYUP SYMMETRICAL ABOUT H'COMB ℄
FLEXIBLE SEAL JOINS METEOROID
SHIELD & STABILITY CONE, FORMING
SEALED PURGE WALL WITHOUT CAPABILITY
OF CARRYING THRUST LOADS
F
.09
.45
.18
.046
GLASS CLOTH REINFORCED
EPOXY BUILD UP AT FRAMES
ISOTROPIC LAYUP
F
.040
.09
1.6
.40
1.5 I.D (REF)
HI-LOK FASTENERS
STRUT
STABILIZING CONE
GLASS CLOTH REINFORCED
EPOXY. ISOTROPIC LAYUP
THICKNESS = .010
ALUMINUM RIBBED FITTING
ENGINE MOUNTING BASE
END BUI
LONGITUD
GLASS CLO
EPOXY END
ISOTROPIC
BOND TO TO

-1



STA 156
.25
GRAPHITE EPOXY EDGE CLOSEOUT BOTH EDGES
1.50 DIA
.60 FLAT FOR FRAME ATTACHMENT
SECTION F-F 2/1
GRAPHITE HOOP FIBERS
.0205 BASIC TUBE WALL
LONGITUDINAL BORON FIBERS (2 MIDDLE LAYERS)
BUILD-UP FIBERS
GITUDINAL BORON
CLOTH REINFORCED
END FITTING (BOTH ENDS)
PIC LAYUP
TO TUBE
GO2 ACCUMULATOR (V = 1.5 CU. FT.)
REF LAYOUT V7-949112
JUNCTION BOX END OF CONDUIT
DOCKING DROGUE
ACPS THRUSTERS
AVIONICS
MODULES (2) CONTAINING TURBO-PUMPS HEAT EXCHANGER & GAS GENERATOR
REF LAYOUT V7-949112
STA 126
STA 156
15°
45° (RADIATOR SPAN)
SECTION E-E 1/2
SYSTEMS OMITTED FOR CLARITY
RADIATOR 4 PLACES (90° SPACING)
REF LAYOUT V7-949113
℄ ORBITER
15°
B
E
A
GH2 ACCUMULATOR (V = 2.4 CU FT) REF V7-949112
FREON PUMP AND ACCUMULATOR
FUEL CELL
ACPS THRUSTERS
DOCKING DROGUE
ENGINE ACTUATOR
REF LAYOUT V7-949
GH2
GO2
ENGINE SUPPLY HELIUM
ACPS THRUSTERS
FOR THERMAL CONTROL PROVISIONS SEE
DWG V7-923604
REF ACPS LAYOUT V7-9491
ELECTRICAL LINE
PNEUMATIC LINES
AVIONICS
REF LAYOUT V7-975403
AVIONICS
ACPS THRUSTERS
PURGE SYSTEM DISCONNECT
LH2 FEEDLINE
DOCKING DROGUE
THIS PANEL (2 PLACES) IS A CONTINUATION OF STABILITY CONE (t=.010) TO AUGMENT TORSIONAL STABILITY OF STRUTS.

F-2



STA 100
STA 126
STA 156
STA 178.5
D
DOCKING DROGUE
D
37°
FITTINGS
949112
THRUST STRUT
LO_2 TANK
THRUST STRUCTURE STABILITY CONE
9112
GRAPHITE EPOXY FRAMES ATTACH TO STRUTS WITH BLIND PULL RIVETS
PURGE WALL & METEOROID SHIELD ATTACH TO FRAME WITH HI-LOK FASTENERS
AVIONICS
SECTION AA
TO STA 126 FRAME
S-GLASS FILAMENT WOUND COMPOSITE STRUTS FOR TANKS & DOCKING DROGUES.
STA 156 FRAME
SECTION DD
STA 126
STA 131.5
GH_2 (V= 2.4 CU FT)
63.5 R
STA 100 ENGINE GIMBAL
LO_2 LINE SEALING BOOT
EXTERNAL THERMAL CONTROL COATING ZINC OXIDE IN RTV METHYL SILICONE
SEALING BOOT
LOCAL DIPT
CAP ALLOWS
MAINTAINING

F-3

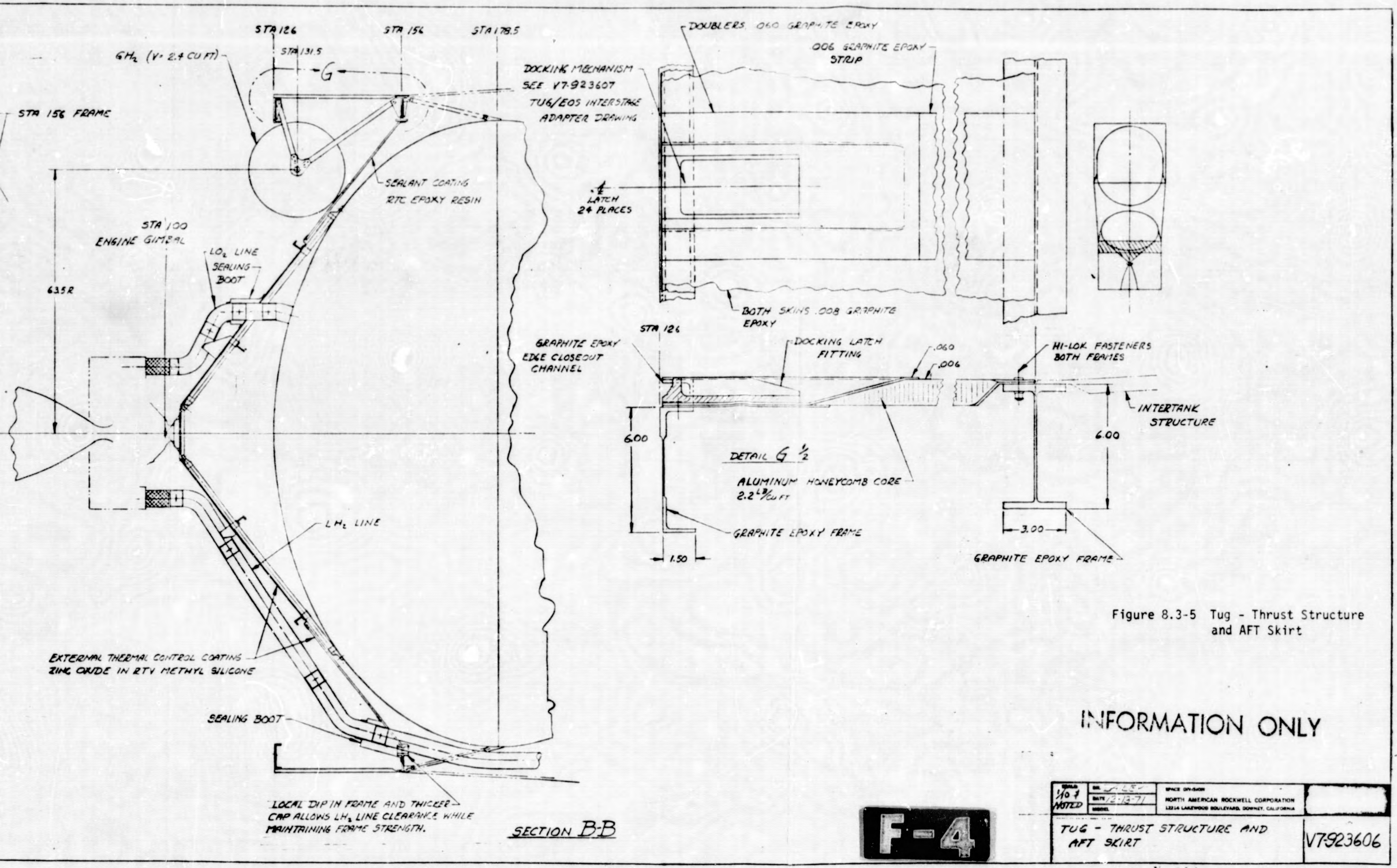


Figure 8.3-5 Tug - Thrust Structure and AFT Skirt

to permit the use of modular manufacturing procedures. Frame splices are located at the edges of the sectors.

The shell functions as a micro-meteoroid bumper to protect the tanks and critical system hardware. It acts as backup for the purge bag and reacts body loads and bending moments. The shell also provides for attachment to both the payload and shuttle adapter using a 24 latch system at each end. Three Shuttle attachment fittings at the forward end of the shell (Station 452) react lateral forces.

The skins of the honeycomb sandwich shell are laminates of high strength (HTS) graphite fiber reinforced composite. This material was selected because of (1) its combined strength and specific modulus over the range of operating temperatures considered, (2) its low thermal expansion coefficient, and (3) the projected range in ply thicknesses available (0.002 inches to 0.008 inches per ply). The skins for all sections are laminates of zero-degree, 90-degree, and 45-degree layers - with the reference axis in the longitudinal direction. These patterns provide the off-axis flexural rigidities for the shell required for local stability, result in a Poisson's ratio of approximately 0.3, and have minimum layup complexities. The zero-degree, 90-degree, and 45-degree layers provide the major portion of the strength and stiffness in the longitudinal, circumferential and shear directions, respectively. For the specific family of laminates selected, a minimum of four layers per skin is required. The minimum skin gage is, thus, 8 mil based on a 2 mil per ply material.

The quarter panel skins for the sandwich shells are laid-up and cured prior to bonding to the honeycomb. Mandrels will be located at the skin surface which will eventually function as the honeycomb interface. This will permit the utilization of minimum weight bonding adhesives by the development of a smooth bond surface. Each skin sector will have thickened edges (doublers co-cured with skin) for the bearing surfaces necessary for mechanical attachment to adjacent sectors and sections. These doublers will be of an isotropic layup for the best combination of bolt bearing, shear-out, and tension strengths. One skin with doubled edge will form the closeout and permit the pinching of the two skins together to closeout the honeycomb. The skins will be bonded to the honeycomb prior to assembly to the frame sectors. The assumed adhesive density was assumed to be 0.04 lb/ft^2/layer, based on current technology.

Aluminum honeycomb was selected for all shell sandwich sections because it can be obtained in lower densities than fiberglass composite core (2.2 lb/ft^3 vs 4.0 lb/ft^3 for 3/16-inch cell). The cell size was set by the skin stability considering cell edge support. Crimping of the honeycomb was used in the check of density requirements.

Mechanical fasteners (Hi-Lok) are used in the design to provide shear load transfer at sector joints and in the joint between sections and major frames. Intermittent mechanical fastening of stability frames will be used as a precaution against peel of the bond between frame and shell. These latter fasteners will penetrate the honeycomb shell at points of locally filled core.

THIS SPACE INTENTIONALLY LEFT BLANK

Major frame sector splices use titanium shims in the caps for bolt bearing capability. The webs at all frame splices will use composite doublers for web shear transfer through mechanical fasteners.

The attachment points for the shell to adjacent components and shell to tank support struts are located at 15 degree stations around the shell. Thus, 24 attachment points are provided to each end of the shell and at the tanks support stations. The points are axially aligned from end-to-end.

The stability and major frames are high strength graphite epoxy composite. The use of this material provides a thermal strain match between shell and frame. The frame webs are laminates of material oriented 45 degrees to the frame axis. This orientation produces maximum shear stiffness and strength. Web layers extend into the caps where they are interspersed with circumferential layers of graphite epoxy. These circumferential layers provide the frame bending rigidity and cap strength. All frames are to be fabricated in four equal sectors, attached to the shell sectors, and spliced together to form a complete circle when the shell sectors are jointed together.

Forward Skirt

The forward skirt is a cylindrical shell as shown in Figure 8.3-4. It extends from Station 304.5 to Station 452 (length equals 147.5 inches). Attachment to the intertank shell occurs at the LH_2 tank support frame by means of a field splice. This skirt contains three I-Section stabilizing frames plus a secondary frame at Station 420.5 for equipment and docking probe support. A C-Section major frame at the forward end of the shell reacts forward lateral Tug-to-Shuttle attachment loads and provides for payload attachment mechanisms at 24 locations.

The skins of the sandwich shell are graphite epoxy. They are each laid-up to produce a quasi-isotropic laminate of four layers - one layer in the longitudinal direction, one layer in the circumferential direction, and the remaining two layers at plus and minus 45 degrees to the longitudinal direction. Each ply of skin material is 2-mil thick. Thus, the total thickness of each skin is 8 mil. The aluminum honeycomb core is 3/8-inch thick.

Cutouts in the honeycomb shell are provided for antennas, an umbilical connection, and for the star tracker and horizon seeker installations. Reinforcement around the cutouts is achieved by adding extra plies to the composite face sheets. For the larger cutouts, channel shaped graphite epoxy intercostals were added to each side. These intercostals run between frames.

The honeycomb sandwich is closed-out along each edge of the quarter panel sectors. This is accomplished by tapering the honeycomb locally along the edge and forming the inner skin to provide the closeout material. Additional composite (doublers) of the same quasi-isotropic pattern as the skins are added locally to each skin to provide additional reinforcement for discontinuity stresses and fastener bearing.

The forward shell frame at Station 452 provides the interface with the payload and with the orbiter cargo bay support structure. Its primary function is to accommodate all orbiter loads (launch, flight, and landing), transmitting them to the Tug shell. The frame has the secondary function of supporting the latching mechanisms that link the payload to the Tug, hence providing the load transfer path between vehicles. The channel-shaped frame is 16 inches deep with triangular web-to-shell gussets under each of the 24 equally-shaped latches. It is attached to the shell by Hi-Lok fasteners. Between the gussets, the web is reinforced by two angle stiffeners. The frame supports the forward end of several avionics component platforms, a TV camera used for docking, and the forward yoke of the docking probe mount. One reinforced cutout to accommodate laser radar has been located in the web. Three external fittings, which terminate in hemispherical ends, are mechanically attached to this frame. These fittings engage the orbiter cargo bay supports. Each fitting is loaded only in the tangential direction to the forward ring frame.

The frame at Station 420.5 provides stability and rigidity to the forward skirt shell. It also supports several avionics component platforms, the rear tubes of the docking probe mounts, and the forward purge pressure-meteor barrier membrane. The 7-5/8-inch deep channel-shaped frame is made of high tensile strength graphite epoxy. The web is reinforced by 72 equally spaced angle stiffeners and is fastened to the shell by Hi-Lok fasteners. Splices occur at the same four equally-spaced locations used for the skirt panel splices, and titanium shims are used in the cap splices to provide an efficient mechanical fastener joint. Two reinforced web cutouts are provided for electrical conduits, and triangular web-to-shell gussets are used to backup the docking probe support tubes.

The three stabilizing frames are also graphite epoxy channels (1-1/2 in. x 3/8 in.). Splices at shell longitudinal joints, however, are accomplished by mechanical fasteners through doubled-up composite rather than through titanium shims, as employed for major frames. These frames are bonded to the inner surface of the shell. Mechanical fasteners are added to localize peel.

For Tug-to-payload docking, a system of three probes, similar in concept to those used for Apollo, are provided. These probes would be greatly simplified from the actual Apollo probes because many of the complex requirements for Apollo-LEM docking would not exist for the permanently fixed probes that would be used for Tug-to-payload docking. Three off-center probes are required because the forward end of the main LH_2 tank is too close to the forward interface plane to allow the installation of a probe assembly on the Tug centerline. Each probe mount consists of an aluminum yoke, 0.10 inch thick, attached to the frame at Station 452, and two graphite epoxy tubes connected to the Station 420.5 frame. The tubes are 2 inch in diameter and 0.036 inch thick. The mounts must sustain simultaneous ultimate loads of 2100 pounds longitudinally and 630 pounds laterally, encountered during docking maneuvers.

Hard latching is accomplished by latching fingers spaced around the periphery of the forward interface. A latch is located at each 15° interval to assure uniform distribution of loading. The latching fingers are translated

by gearing and are electrically actuated. Redundancy can be achieved by use of dual, parallel drive motors or by use of override pyrotechnic devices.

All equipment is located forward of the pressure barrier. Access is from the open, forward end of the Tug. An umbilical connection of the Tug to the orbiter cargo bay is located in the forward skirt for the LH_2 fill-and-drain, pressurization, and electrical lines. Avionics equipment is mounted on eight rectangular, aluminum honeycomb panels which are supported in the forward skirt between the two forwardmost ring frames (Stations 420.5 and 452). To contain purge gas inside the structural shell, a spherically contoured diaphram of rubber impregnated glass cloth is attached near the inboard cap of the ring frame at Station 420.5. This diaphram also serves as a meteoroid barrier. The LH_2 fill and drain line passes through a sealed, elliptical cutout in this diaphram. To avoid interference of the star tracker and horizon tracker installations with the Tug-to-Shuttle attach fittings the star tracker-horizon tracker axis is rotated 30 degrees from the orbiter Z axis. The Z axis of the Tug lies on the star-tracker-horizon tracker axis.

Intertank Shell

The intertank shell is a cone frustrum spanning between the LOX and LH_2 tank support. The shell, which is detailed in Figure 8.3-5, is 148.5-inch long and varies in radius from 81 inch at the aft end to 90 inch at the forward end. The caps of the LOX tank support frame (Station 156) and the LH_2 tank support frame (Station 304.5) act as a splice plate for the field joining adjacent shell section. Exterior thermal insulation is shown in the areas of the APS plumes to protect the sandwich shell.

The sandwich shell employs graphite epoxy face sheets and 0.7 inch thick aluminum honeycomb core. The face sheet thickness and layup pattern are identical to the face sheet previously described for the forward skirt. Thus, each face sheet is 8-mil thick. As described for the forward skirt, the sandwich honeycomb core is closed out along all edges of the quarter panels by the skins and composite doublers.

An access door of 30-inch square opening is located in the shell to allow installation of equipment into the LOX tank and provide access to the area between the LH_2 and LOX tanks. This door is non-structural. Body loads are sheared to the sides of the door cutout by means of skin doublers and are carried by intercostals along the door edges. These intercostals span between adjacent frames.

The LOX tank support frame at Station 156 is an I-Section, 6-inch deep by 3-inch wide. This graphite epoxy frame is to be spliced at quarter panel positions using Hi-Lok fasteners through titanium shims locally positioned in the composite material. This frame provides twenty-four equally spaced, concentrated load point fittings for attachment to the forty-eight LOX tank support struts. The struts and tank of Figure 8.3-4 are discussed in another section. This frame is mechanically attached to the sandwich shell with Hi-Lok fasteners.

The LH_2 tank support frame is located at the forward end of the intertank shell (Station 304.5). This frame is similar to the LOX tank support frame, but smaller. Its dimensions are 3.7 by 1.0 inches. This frame, in addition to providing LH_2 tank support, also provides, through one-half of its outer cap, a splice plate for the mechanical attachment to the forward skirt. The same number of tank support strut attachment points (24) are provided as for the LOX tank support frame. The frame is attached to the shell by Hi-Lok fasteners.

Three stability frames are located at approximately equally spaced positions between the two tank support frames. These frames are bonded to the inner surface of the sandwich shell quarter panels. Mechanical fasteners are also used to tie these frames to the shell. The main purpose of these fasteners, however, is to localize any possible peel between shell and frame. Therefore, the fasteners are positioned at greater distance than would be used if the fasteners were the primary load carrying component. The honeycomb core is locally filled for each fastener. An umbilical connection to the orbiter cargo bay for LOX fill and drain and pressurization and electrical lines is located in the intertank structure. Venting provisions will be incorporated into the intertank area to vent the pressurized portion of the shell structure and to allow rapid evacuation of the multilayer insulation.

Aft Skirt

The cylindrical aft skirt is shown in Figure 8.3-5. It is 30-inches long and has a 81-inch radius. This sandwich shell has a major frame at the aft end (Station 126) and attaches to the LOX tank support frame of the intertank shell (Station 156) by a field joint. This structure houses the orbiter-to-Tug docking drogues, engine systems equipment, and some avoinics equipment. In addition, support for the aft micrometeoroid protection shield and purge bag support cone is provided. A series of 24 latches are provided at the aft end for coupling the Tug to the Tug/Shuttle adapter.

The sandwich shell of the aft skirt uses graphite epoxy skins and 1.13 inch thick aluminum honeycomb core (2.2 lb/ft^3 density). Each face sheet is the equivalent of five layers thick (10 mil per skin). The four layers of composite material next to the core are uniformly distributed over the total area of the shell. They are arranged in the quasi-isotropic pattern previously described for the forward skirt and intertank shell. The outer layer of each face sheet is not uniformly distributed. Each outer layer is stacked on itself to form local thickened longitudinal areas that act as longerons between the 24 latches and 24 LOX tank support points. The stacked areas of material are eight inches wide and three layers thick (6 mil).

The edges of the shell quarter panels are closed-out in the same manner as described for the forward skirt and intertank shell; except for the aft end. This edge uses a composite "C" channel.

The 24 latches for coupling to the Tug/Shuttle adapter are recessed into the sandwich to minimize the discontinuity bending moments. These recesses are local and are achieved by an integral box and skin doubler system that is secondarily bonded to the outer face sheet. A doubler is also positioned on

the inner face sheet at latch attach points. Graphite epoxy laminates were used for these doublers. These laminates were composed primarily of material at 45 degrees to the longitudinal direction for maximum shear rigidity. Titanium shims were located within the composite in the areas where the latches bolted to the shell in order that sufficient bearing could be achieved.

The aft end of the shell (station 136) is the mating plane of the Tug and Shuttle adapter. Two identical frames are located at this station - one on the aft skirt and one on the adapter. These channel frames maintain the shape of the shells during mating and react radial loads developed under latch loads. Each frame is 5-1/2 inches deep and has 1-1/2 inch wide caps. Triangular web to shell gussets are located under each of the 24 equally spaced latches. Between gussets the frames are reinforced by two angle stiffeners. Cap splices in these frames employ titanium shims. Web splices require only build-up of the composite.

Adapter

The Tug-to-Shuttle adapter is an 81-inch long cylinder with an 81-inch radius (see Figure 8.3-6). It extends from Station 45 to Station 126 and stays with the Shuttle during Tug operation. Its primary function is to provide attachment to the orbiter, accepting concentrated loads at the Shuttle interface and shearing them out to a near uniform load at the interface with the Tug interface. The Shuttle cargo bay structure provides two-point support for axial and vertical loads and a single support point for lateral loads. Bottles for storage of helium purge gas are located within the shell, as are Tug docking probes. Major frames are located at each end of the shell. Two stability frames are also used.

Two pair of titanium plates act as longerons to shear concentrated axial loads into the shell. Each pair is diametrically opposite the other. One of each pair is located in each face sheet of the honeycomb sandwich shell. These titanium longerons provide bolt bearing for the connection of Shuttle attachment hardware. They are tapered in width and thickness in the axial direction to control the strain as the load is sheared out. The longeron is 56-inches long and extends from Station 45 to Station 101. The edge of the metal adjacent to the composite skins is chem-milled to form a step taper. This taper provides for the bond of graphite epoxy layers to the titanium for the structural tie between shell and longeron. This bond is to be accomplished when the composite skin is cured so that the sector of skin and the longeron of each face sheet are one unit after composite cure. Titanium was selected for this component because of its high bearing strength and low thermal expansion coefficient which more nearly matches that of the composite than steel or aluminum.

The shell utilizes graphite epoxy face sheets with a 1.12-inch thick aluminum honeycomb core. These face sheets are divided into three major areas - an area of low load, an area of load transition from high to moderate load, and an area of high shear load. The low load section is located in the region where the loads have not been sheared into the shell and are carried primarily by the longerons. This area is composed of four layers of graphite epoxy patterned to form a quasi-isotropic laminate similar to the skins of the forward skirt. The skin thickness in this area is 8 mil.

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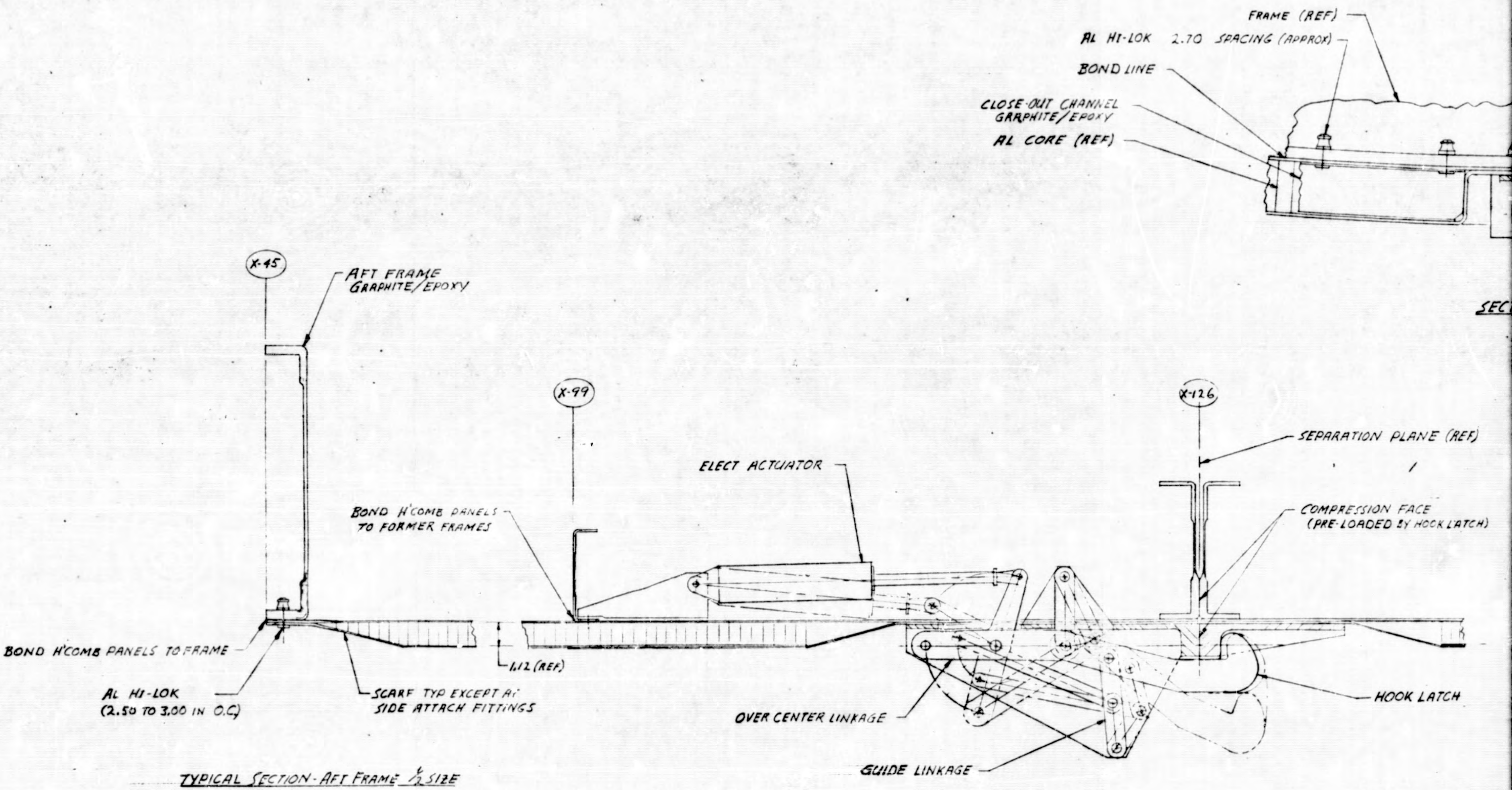
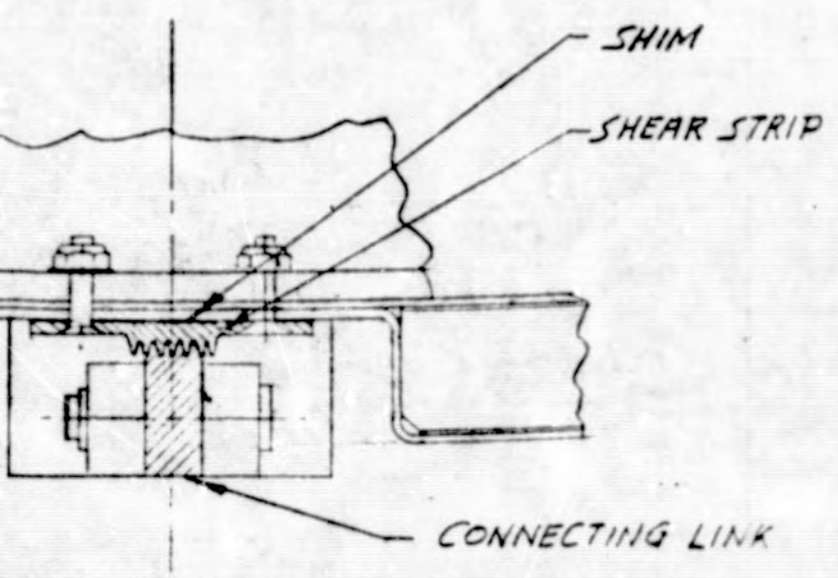
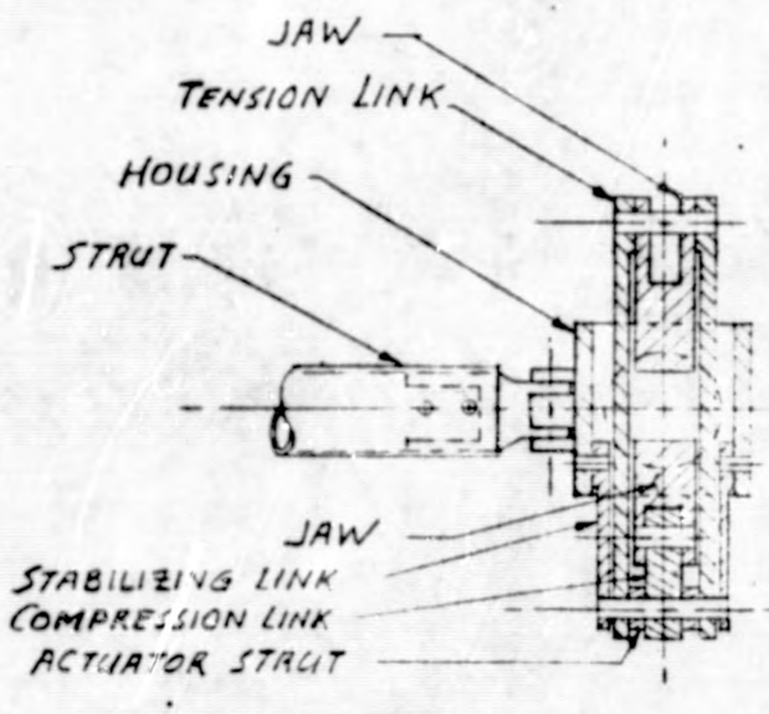

FRAME (REF)
AL HI-LOK 2.70 SPACING (APPROX)
BOND LINE
CLOSE-OUT CHANNEL
GRAPHITE/EPOXY
AL CORE (REF)
SECT
X-45
AFT FRAME
GRAPHITE/EPOXY
X-99
X-126
SEPARATION PLANE (REF)
ELECT ACTUATOR
BOND H'COMB PANELS
TO FORMER FRAMES
COMPRESSION FACE
(PRE-LOADED BY HOOK LATCH)
BOND H'COMB PANELS TO FRAME
1.12 (REF)
AL HI-LOK
(2.50 TO 3.00 IN O.C)
SCARF TYP EXCEPT AT
SIDE ATTACH FITTINGS
OVER CENTER LINKAGE
HOOK LATCH
GUIDE LINKAGE
TYPICAL SECTION - AFT FRAME 1/2 SIZE
SECTION FF 1/2 SIZE (ROTATED)
TUG/ADAPTOR LOCK/REL LATCH (4 PLCS)

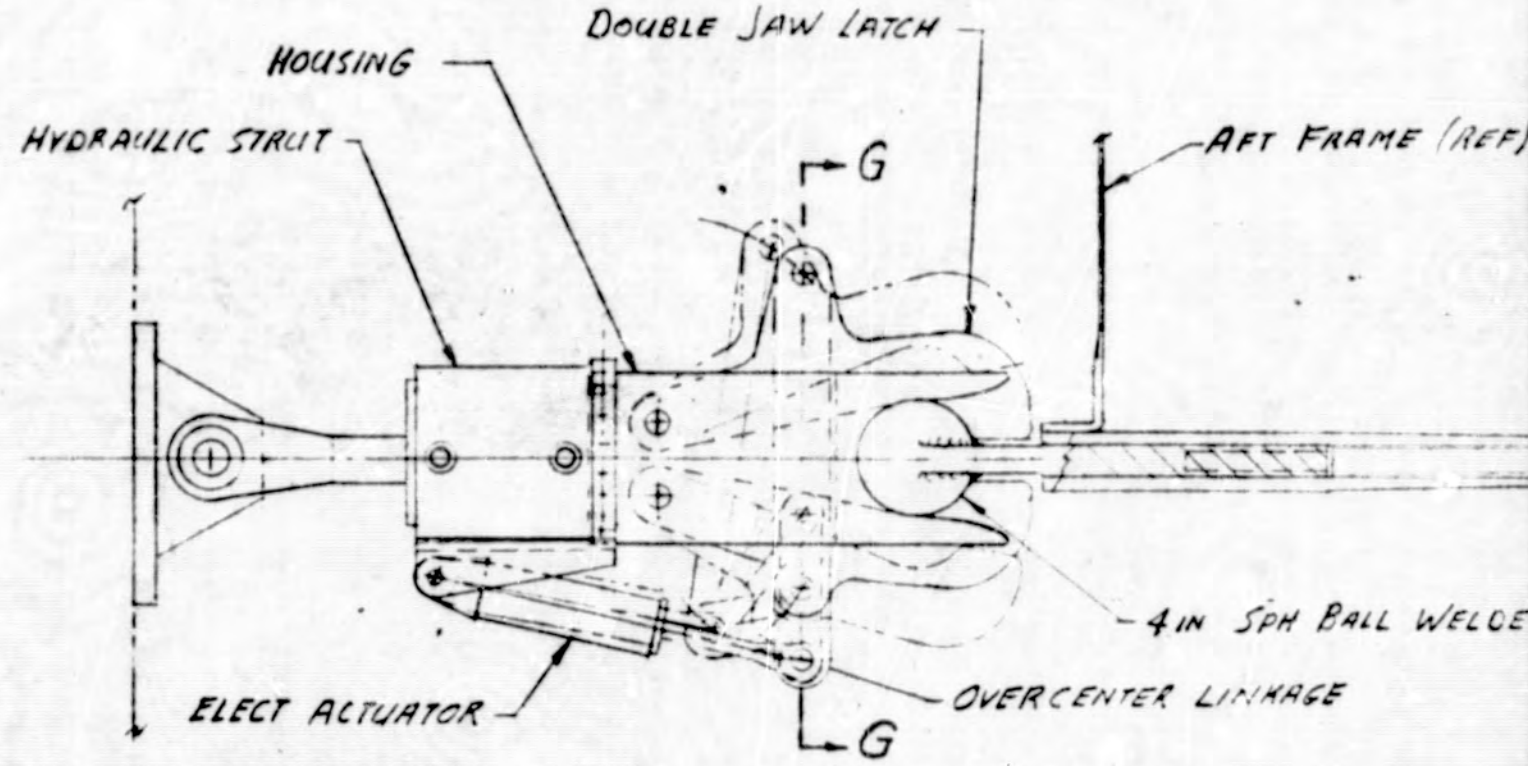
F-1



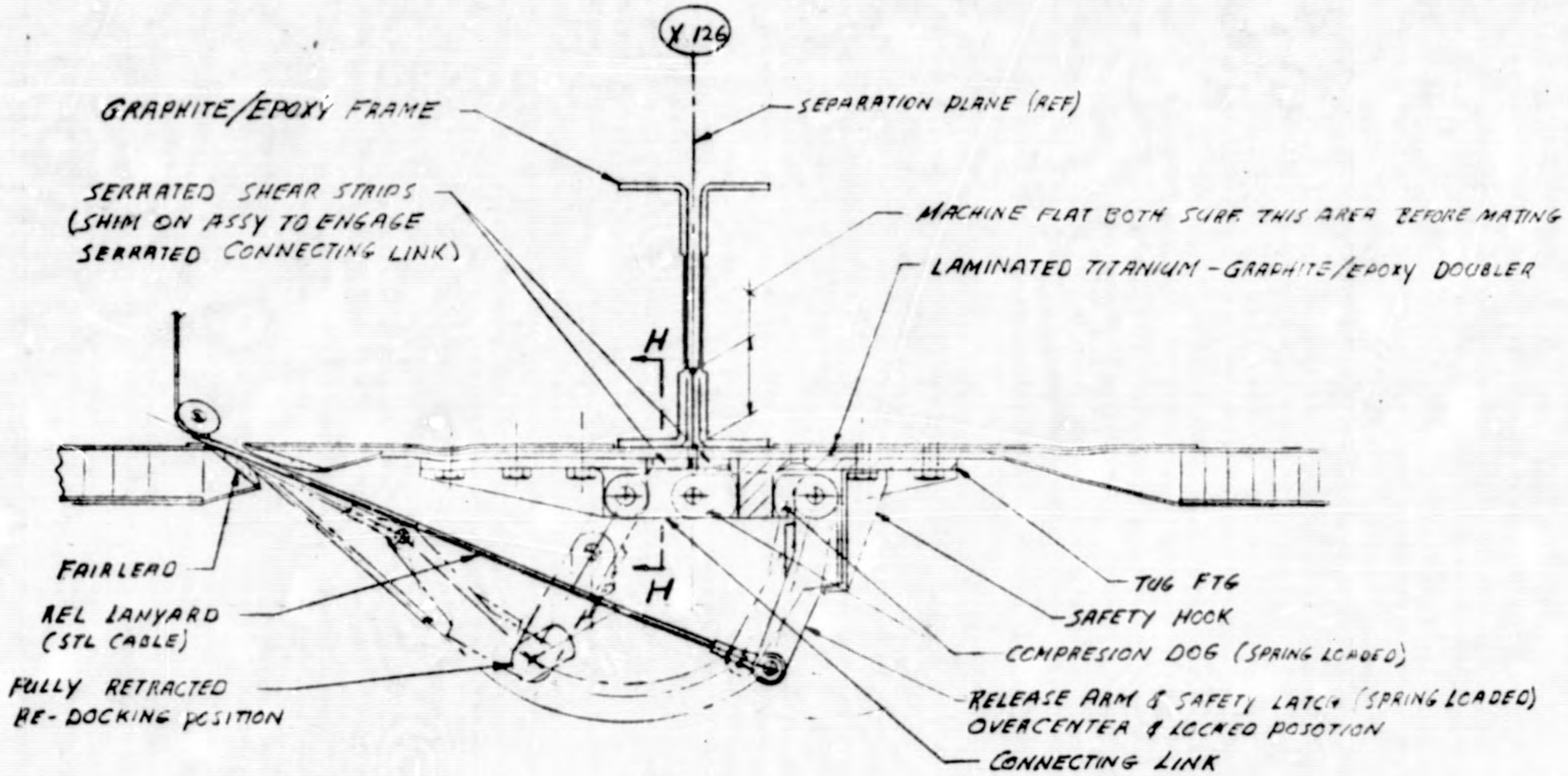
ECTION HH FULL SIZE



SECTION GG 1/4 SIZE

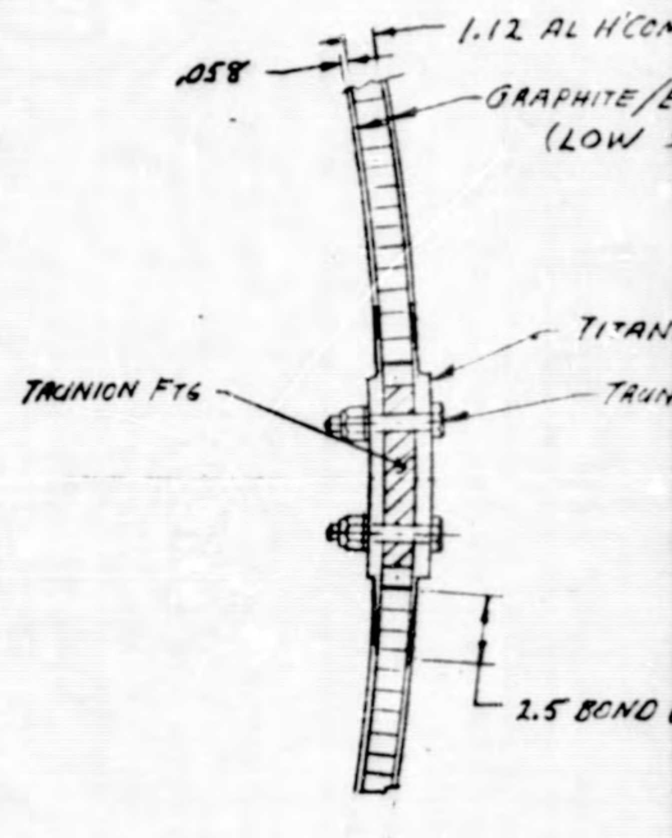


SECTION DD 1/4 SIZE



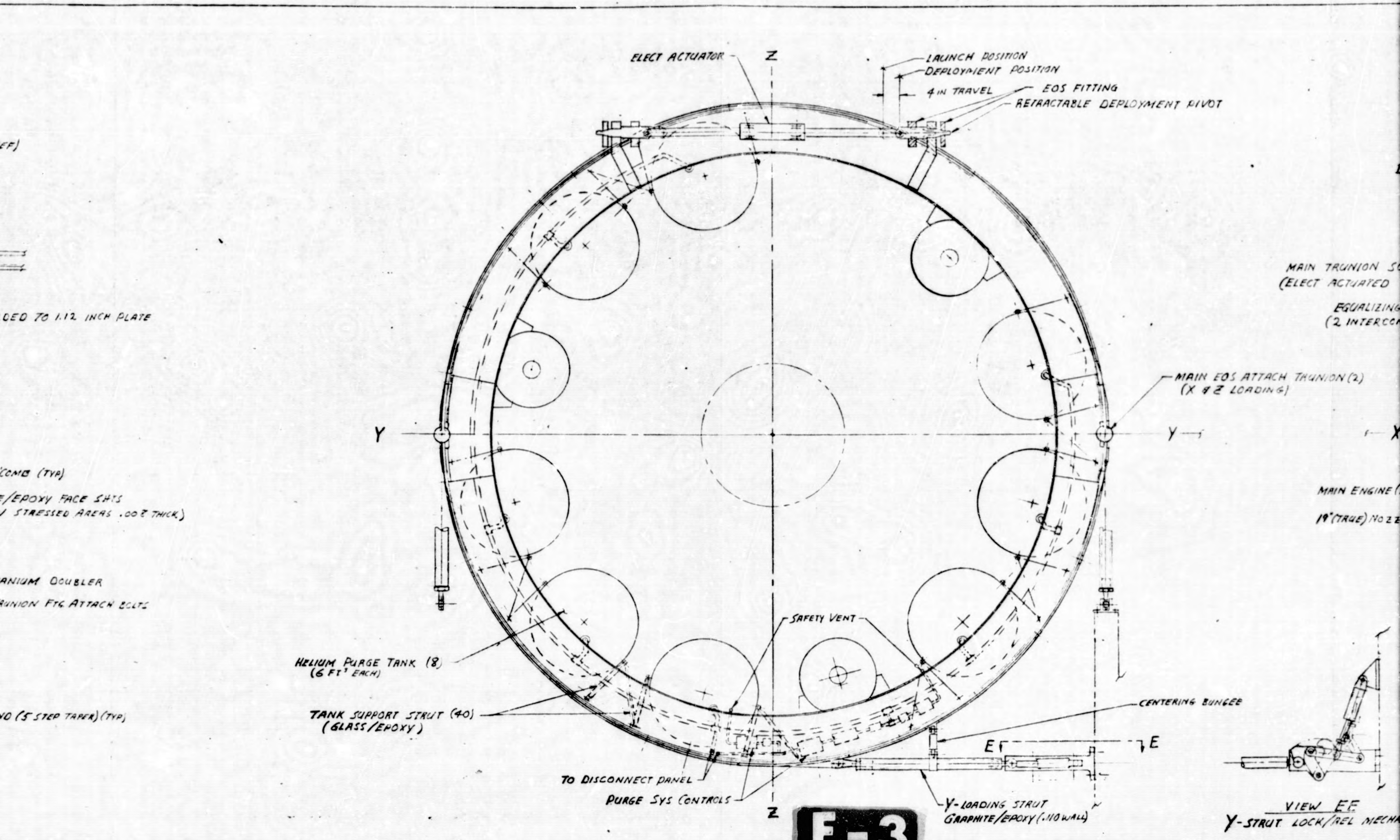
DETAIL B 1/2 SIZE

REQUIRED FOR LAUNCH LOADS ONLY
INACTIVE FOR DOCKING & ENTRY

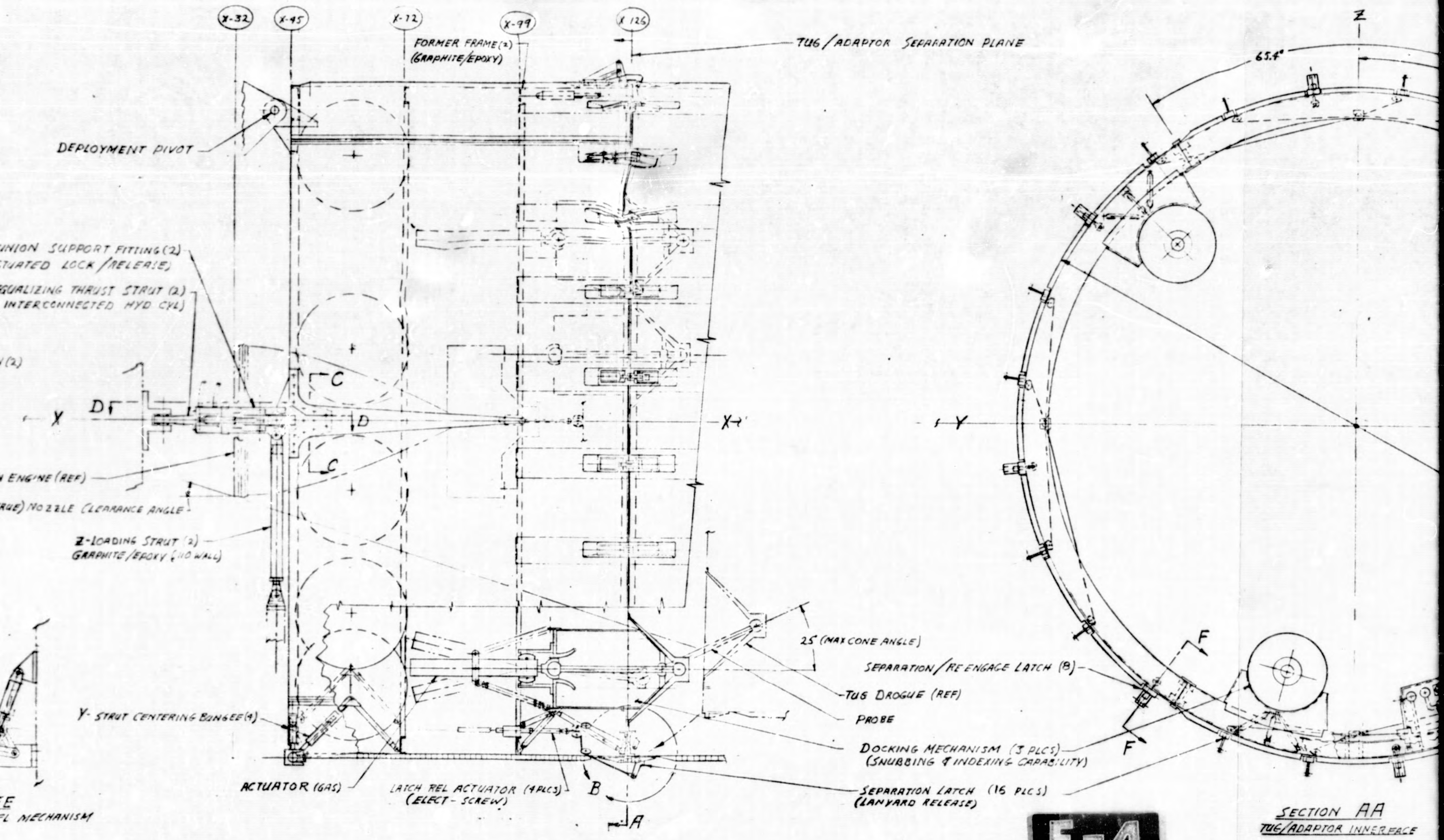


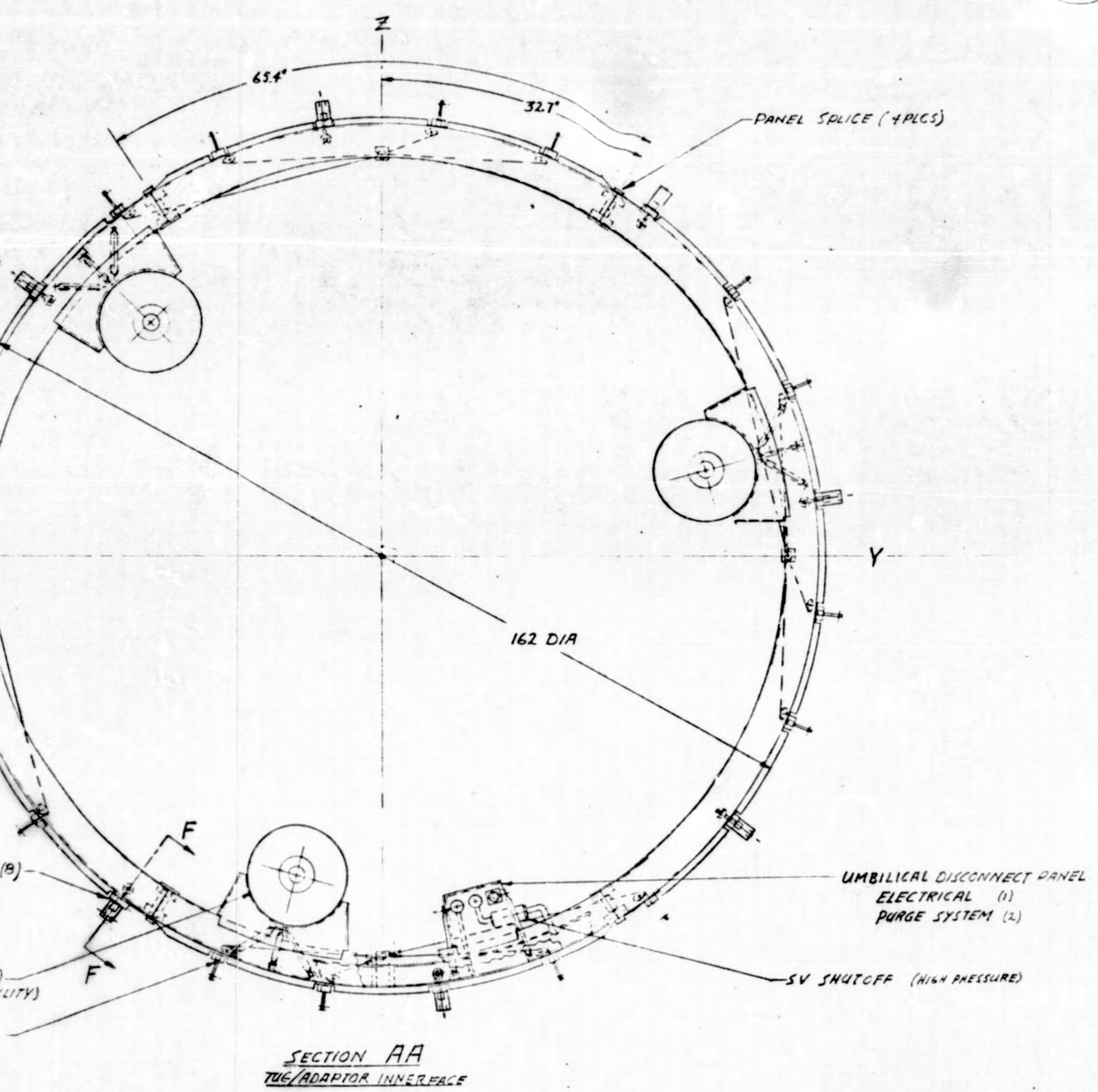
SECTION CC 1/4 SIZE

F-2


ELECT ACTUATOR
Z
LAUNCH POSITION
DEPLOYMENT POSITION
4 IN TRAVEL
EOS FITTING
RETRACTABLE DEPLOYMENT PIVOT
EF)
DED TO 1.12 INCH PLATE
MAIN TRUNION S
(ELECT ACTUATED
EQUALIZING
(2 INTERCON
MAIN EOS ATTACH TRUNION (2)
(X & Z LOADING)
Y
Y
X
COMB (TYP)
E/EPOXY FACE SHTS
STRESSED AREAS .002 THICK)
MAIN ENGINE
19" (TRUE) NOZZ
ANIUM DOUBLER
RUNION FTG ATTACH BOLTS
SAFETY VENT
HELIUM PURGE TANK (8)
(6 FT³ EACH)
NO (5 STEP TAPER) (TYP)
TANK SUPPORT STRUT (40)
(GLASS/EPOXY)
CENTERING BUNGEE
E
E
TO DISCONNECT PANEL
PURGE SYS CONTROLS
Z
Y-LOADING STRUT
GRAPHITE/EPOXY (.110 WALL)
VIEW EE
Y-STRUT LOCK/REL MECHA

F-3


X-32
X-45
X-72
X-79
X-126
Z
FORMER FRAME (2)
(GRAPHITE/EPOXY)
TUG / ADAPTOR SEPARATION PLANE
65.4°
DEPLOYMENT PIVOT
UNION SUPPORT FITTING (2)
ACTUATED LOCK / RELEASE)
EQUALIZING THRUST STRUT (2)
INTERCONNECTED HYD CYL)
(2)
D
C
Y
X
ENGINE (REF)
(LARGE) NOZZLE CLEARANCE ANGLE
Z-LOADING STRUT (2)
GRAPHITE/EPOXY (.110 WALL)
25 (MAX CONE ANGLE)
SEPARATION / RE ENGAGE LATCH (8)
F
TUG DROGUE (REF)
PROBE
Y-STRUT CENTERING BUNGEE (4)
DOCKING MECHANISM (3 PLCS)
(SNUBBING & INDEXING CAPABILITY)
ACTUATOR (GAS)
LATCH REL ACTUATOR (4 PLCS)
(ELECT - SCREW)
B
SEPARATION LATCH (16 PLCS)
(LANYARD RELEASE)
A
EL MECHANISM
SECTION AA
TUG/ADAPTOR INTERFACE
F-4



INFORMATION ONLY

Figure 8.3-6 Adapter Structure - Tug to Shuttle Orbiter

1:10 NOTED		SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
ADAPTER STRUCTURE - TUG TO SHUTTLE ORBITER			V7-923607

F-5

The high shear area is bonded to the longerons and extends at an angle of 45 degrees from the forward and aft ends of the longeron. The material in this area is 70 percent ± 45 degree material for high shear rigidity. This property permits the shear out of axial loads with minimum shear distortion of the shell. This area is 58 mil thick per skin with a mix of 10 percent and 20 percent circumferential and longitudinal layers, respectively.

The transition area is triangularly shaped. It extends from the tip of the longeron (Station 101) to the Tug interface (Station 126). This area provides for longitudinal deformation as an axial continuation of longeron deformation. This deformation is controlled by section stiffness to match the combined shear and longitudinal deformations of circumferentially adjacent areas. In the vicinity of the longeron, the area is the same thickness as the high shear area. The thickness reduces in proportion to the axial distance from the longeron - to a thickness of 10 mil at the Tug interface. Reduction in thickness is accomplished by the elimination of face sheet plies.

The adapter provides for attachment to the aft skirt of the Tug by including 24 latches. These latches are recessed into the honeycomb shell in a manner similar to the technique used for the recessed latches of the aft skirt.

The adapter is divided into four sectors for manufacturing convenience. The longitudinal joint for these sectors was positioned so that it did not pass through the highly loaded area of the shell. Therefore, it passes only through the minimum gage area of the skin. Because of this desire to locate outside the high stress area, the four sectors of the shell were not equal. Two suspended arcs of 115 degrees and two suspended arcs of 65 degrees each. Sector splices were accomplished in the manner described for the Tug shell.

The adapter has two major and two stabilizing frames. The frame at Station 45.0 provides the aft interface with the orbiter cargo bay support structure. Its primary function is to accommodate all orbiter loads (launch, flight, and landing), transmitting them to the adapter and Tug. The channel-shaped frame is 12 inches deep with the web reinforced by 72 equally-spaced angle stiffeners. The adapter frame at Station 126 is identical to the aft skirt frame at the same station. The stabilizing frames are each four inches deep by one inch wide. One of the frames is located at the top of the longeron (Station 101) and the other halfway down the longeron (Station 73).

The launch thrust load is transmitted to the adapter structure through two interconnected, equalizing hydraulic struts. Pitch and yaw loads are introduced tangentially through external fittings at the sides and bottom. Axial load attach points are in line with the neutral axis of the honeycomb shell to minimize eccentricities and bending moments. For Tug deployment, a shaft near the upper forward corner of the Shuttle cargo bay is inserted into receiving fittings on the adapter. The adapter Tug combination may then be rotated out of the cargo bay.

The heel lines of the aft skirt and adapter frames at Station 126 form the Tug adapter interface. Twenty-four latches are located externally around the perimeter of the interface. Twenty of the twenty-four latches are folded

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away after initial deployment leaving only the remaining four, independently actuated latches for use for redocking and reentry when only Tug dry weight loads would be encountered. Redocking is accomplished by the engagement of three probes into drogues housed in the Tug aft skirt area. When extended the probe arms are free to deflect laterally under slight pressure. When the probes have drawn the Tug, aligned and into contact with the adapter interface, the four reentry latches are actuated. The adapter-Tug combination is rotated back into the cargo bay. The attachment fittings are engaged and the deployment shaft withdrawn.

8.3.2 Internal Body Shell Loads

The Tug vehicle was analyzed for the external body loads and Space Shuttle Orbiter attachment loads, shown in Section 8.2. The internal loads solution was accomplished using the NARSAMS Structures Analysis Computer Program. This program is a modification of ELAS-A, which was developed for NASA by Jet Propulsion Laboratory. A complete description of ELAS-A is given in Reference 6. NARSAMS is a general-purpose digital computer program for the in-core solution of linear equilibrium problems of structural mechanics. The finite element stiffness method is used for the solution. Almost any geometry and structure may be handled because of the availability of linear, triangular, quadrilateral, tetrahedral, hexahedral, conical, and triangular and quadrilateral torus elements. Stresses and loads are provided by the best-fit strain tensors in the least-squares sense at the model node points where the deflections are given. The number of unsuppressed degrees of freedom that can be handled in a given problem is 500 to 600 for a typical structure, which is sufficient to model the complete Tug vehicle.

The input to NARSAMS consists of:

1. Structural model node point locations in the overall coordinate system (X, Y & Z).
2. The elastic properties of the finite elements (modulus of elasticity, moments of inertia, area, thickness and shear modulus).
3. Applied external loads and boundary conditions.

The program output consists of:

1. The node point deflections.
2. Internal loads at the node points.
3. Stresses in the finite elements.
4. Tug to Orbiter attach point reactions.

The Tug structural model is shown in Figure 8.3-7. Because of symmetry, only one-half of the structure is modeled. The model consists of simulated payload outer shell structure, Tug body shell structure, and the Tug to Shuttle adapter structure. In the model, the axial shell structure is lumped into

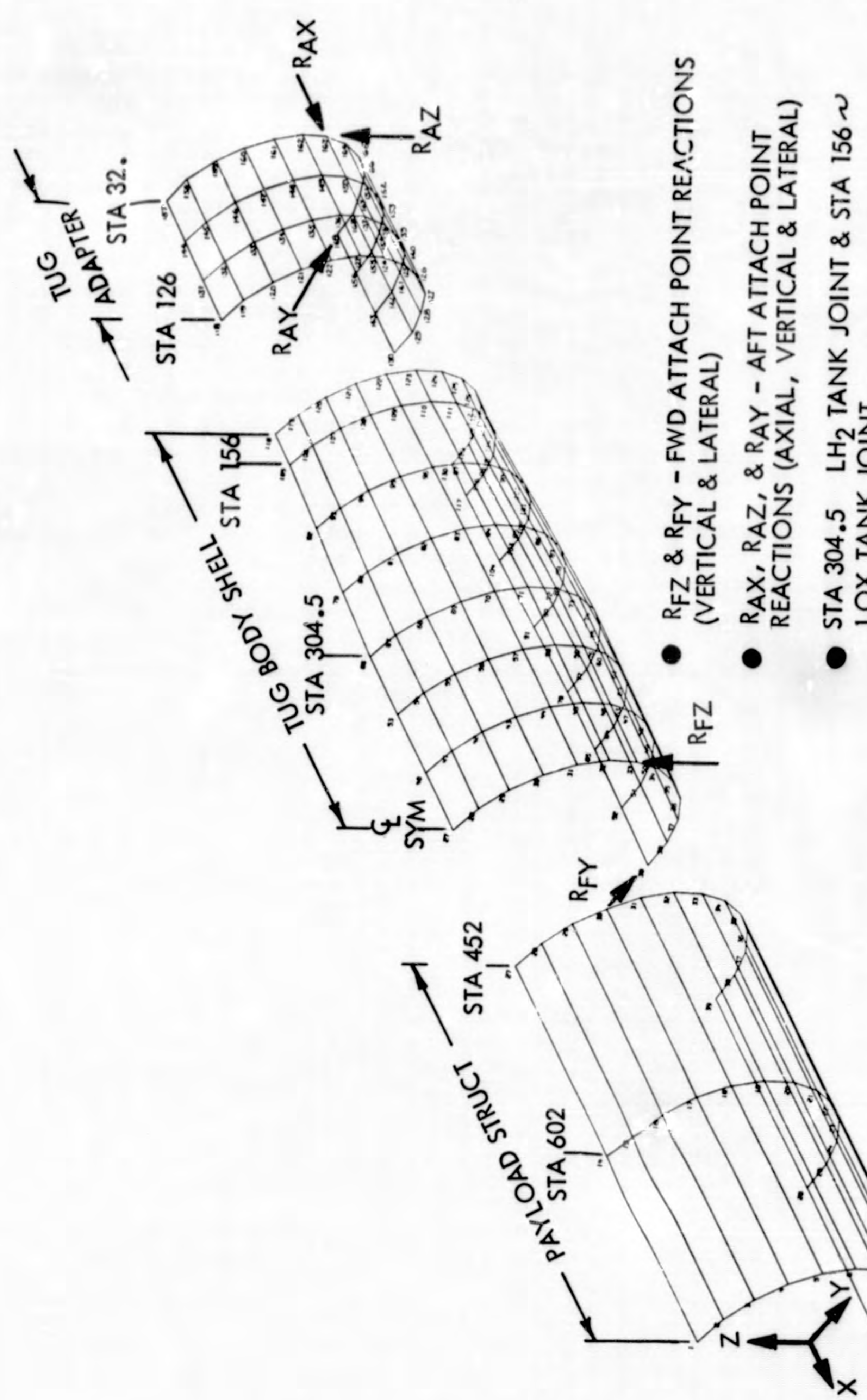

Figure 8.3-7 Tug Internal Loads Calculations Structural Model

twenty-four longerons connected at the node points. The frames are represented as beam elements between node points. The longerons and beam elements are connected by shear panels that are set to match the shear stiffness of the shell structure. The external load distribution is applied as concentrated loads at the nodes at stations 602, 304.5, and 156. Station 602 is the mid-span of the payload. Stations 304.5 and 156 are the LH_2 tank and LOX tank attachments to the shell structure, respectively.

The solution involves a process of iteration. The structure is rough sized using a simple $Mc/_I$ plus $P/_A$ distribution of the external loads. This "rough sized structure" is then used as input data in the first internal loads computer run. The internal loads output from the first run is used to resize the structure, which is then verified by a second computer run. The loads shown in this section are from the second run. Figures 8.3-8 and 8.3-9 show the maximum shell load intensity (Nx). This is shown as a lengthwise distribution from station 156 on the aft skirt to station 752, (the forward end of the payload) on Figure 8.3-8, and as a circumferential distribution at three stations on the adapter and at mid-span of the first bay of the aft skirt on Figure 8.3-10. This circumferential plot is used to illustrate the non-uniform loading in the adapter caused by the concentrated attach fitting loads. The critical load condition for all but a small portion of the forward skirt (- Nx) is the -3.0 G launch condition for 56.94 kip Tug weight and 8.06 kip payload. Figures 8.3-10 and 8.3-11 delineate the internal loads of the attachment backup frames at stations 45 and 452. These are the only frames in the Tug structure that are loaded heavily. The -3.0G launch condition is critical for these frames. However, the station 45 frame is critical at 62.0 kip Tug weight with no payload, and the station 452 frame at 56.94 kip Tug weight with the 8.06 kip payload.

In the analysis, all of the external load conditions shown in Section 8.2 are applied. The program computes the internal loads and performs a search for maximum values. The program printout then contains these maximum internal loads with the corresponding critical load condition indicated. A listing of the load conditions investigated follows:

1. Launch at axial accelerations of -3.0G for initial -3.0G during first stage boost, and +0.2G for dynamic release. Each condition was investigated for a 56.94 kip Tug weight with a 8.06 kip payload, and a 62.0 kip Tug weight with no payload.

2. End first stage boost at an axial acceleration of -3.3G with the same Tug/payload combinations indicated for Item 1.

3. Orbiter re-entry at a 6.36 kip Tug weight with a 4.16 kip payload. This condition yields the maximum Z axis acceleration of +4.0G's.

4. Orbiter landing at accelerations of +1.3G axial and +3.2 G Z axis, with the 6.36 kip Tug weight and 4.16 kip payload. This represents a hard landing with an empty tug and the maximum return payload.

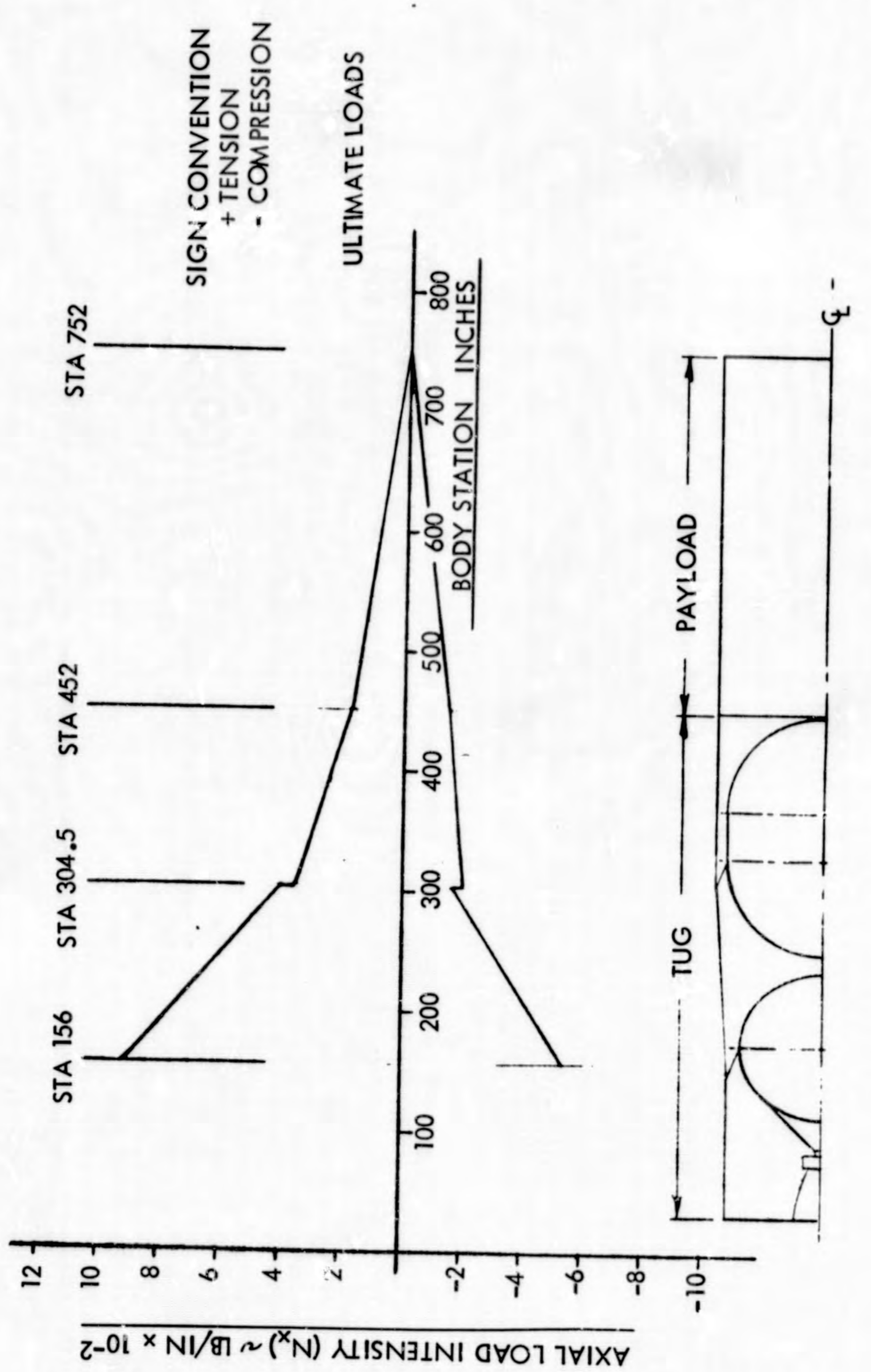

Figure 8.3-8 Internal Loads Maximum Body Shell Axial Load Intensity

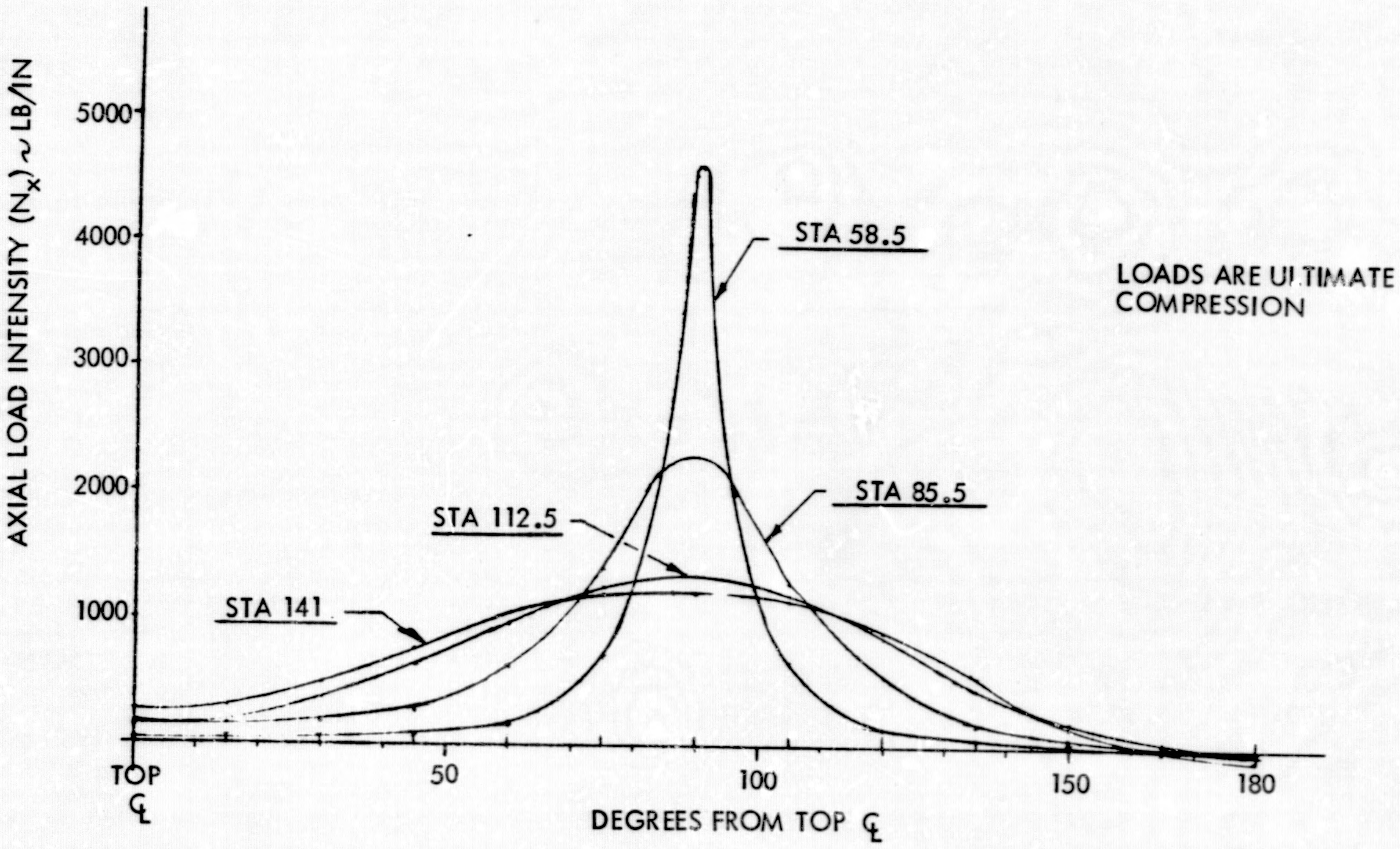

Figure 8.3-9 Internal Loads Adapter and Aft Skirt Maximum Axial Load Intensity

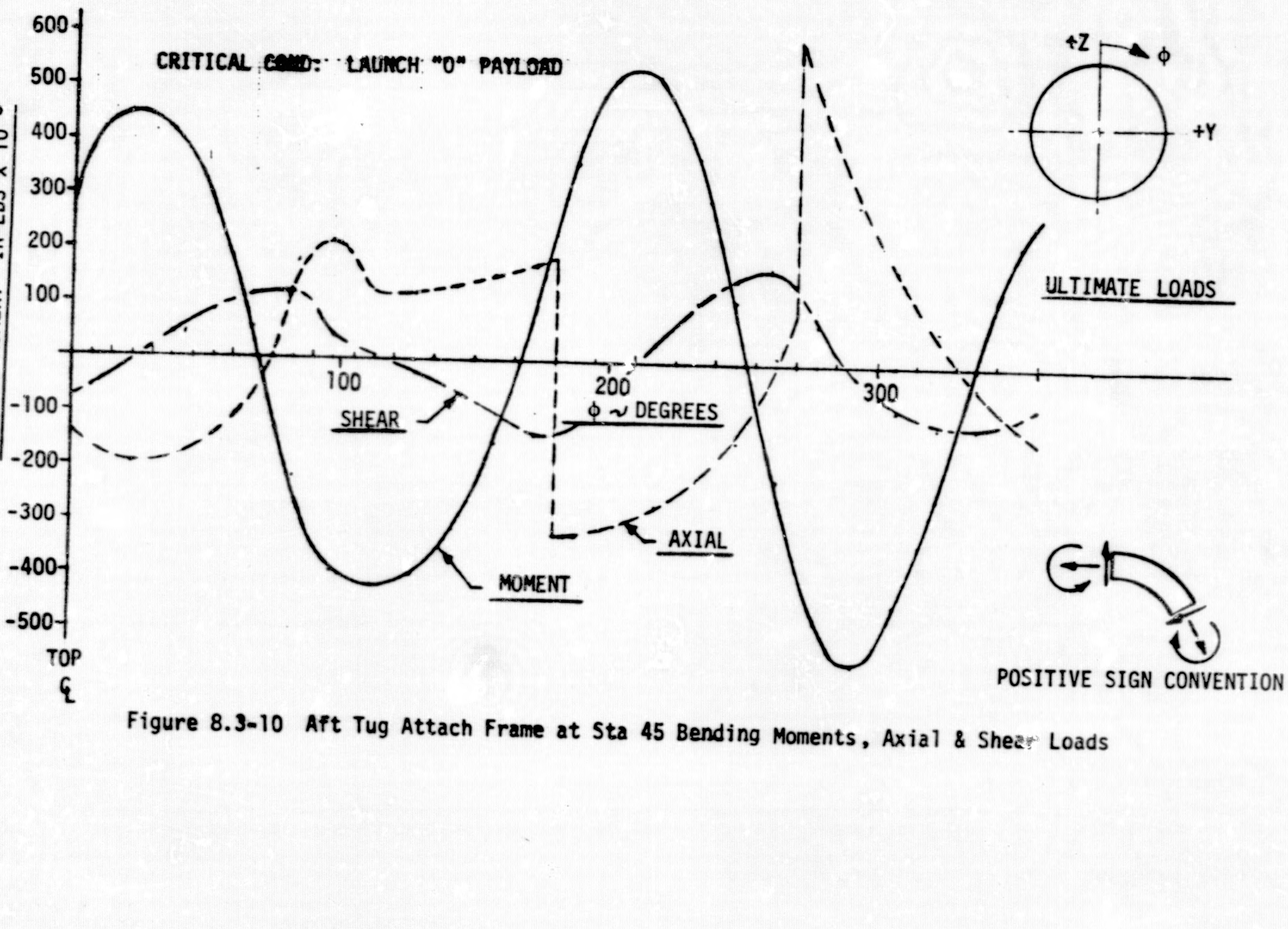

Figure 8.3-10 Aft Tug Attach Frame at Sta 45 Bending Moments, Axial & Shear Loads

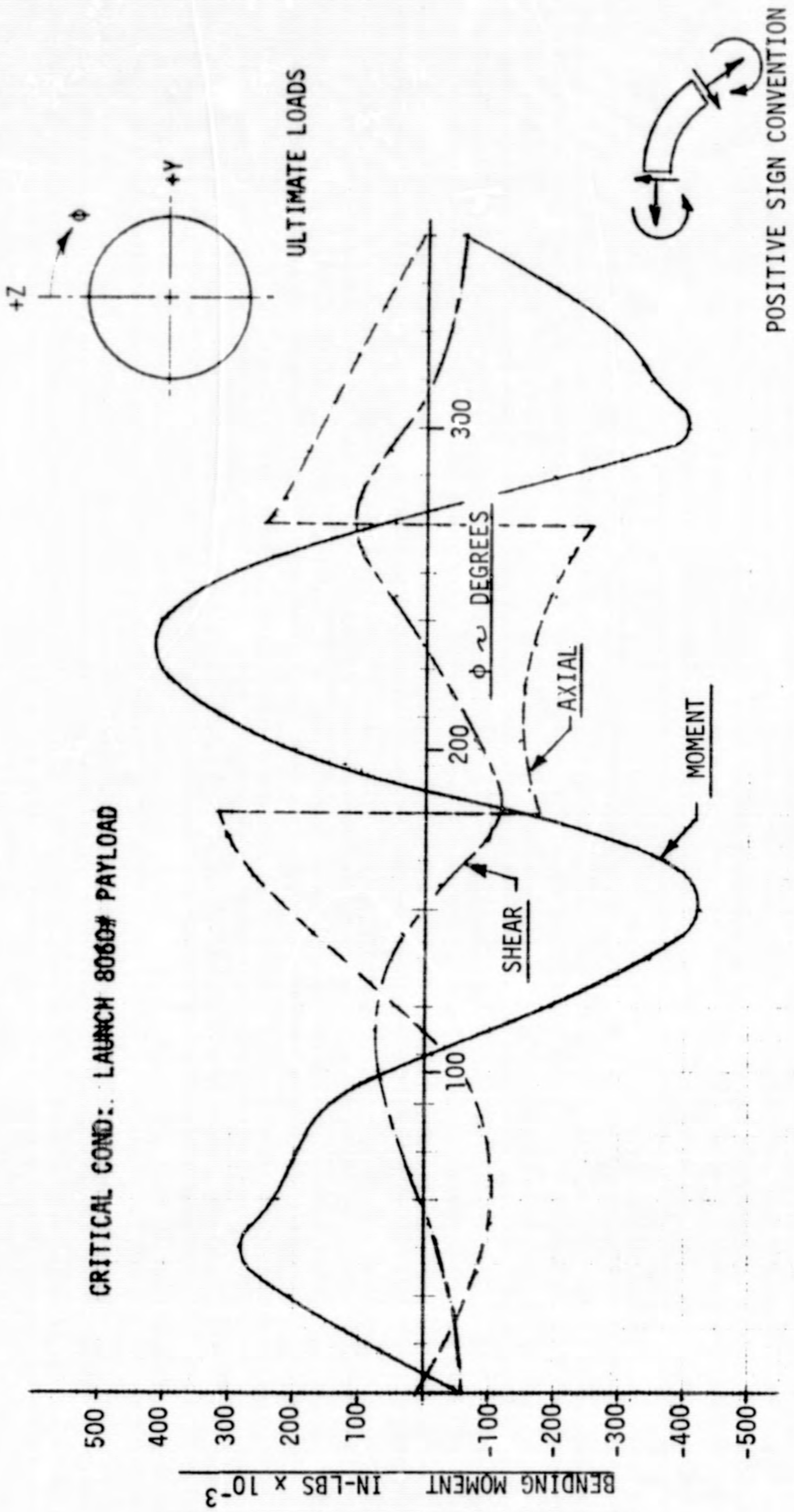

Figure 8.3-11 Fwd Tug Attach Frame at Sta 452 Bending Moments, Axial & Shear Loads

The maximum Tug and payload body deflections due to the internal loads shown in the section are delineated in Table 8.3-1. The maximum payload deflections occur at the Tug/payload interface and are influenced by the forward Tug support frame (Station 452). The Tug maximum body deflections occur at Stations 304.5 and 200 for the lateral and normal motions, respectively. The -3.0G, 8.06 kip payload launch conditions generates all of these deflections. Deflections due to dynamic loadings are not available at this time because the establishment of vibrational environments is beyond the scope of this analysis.

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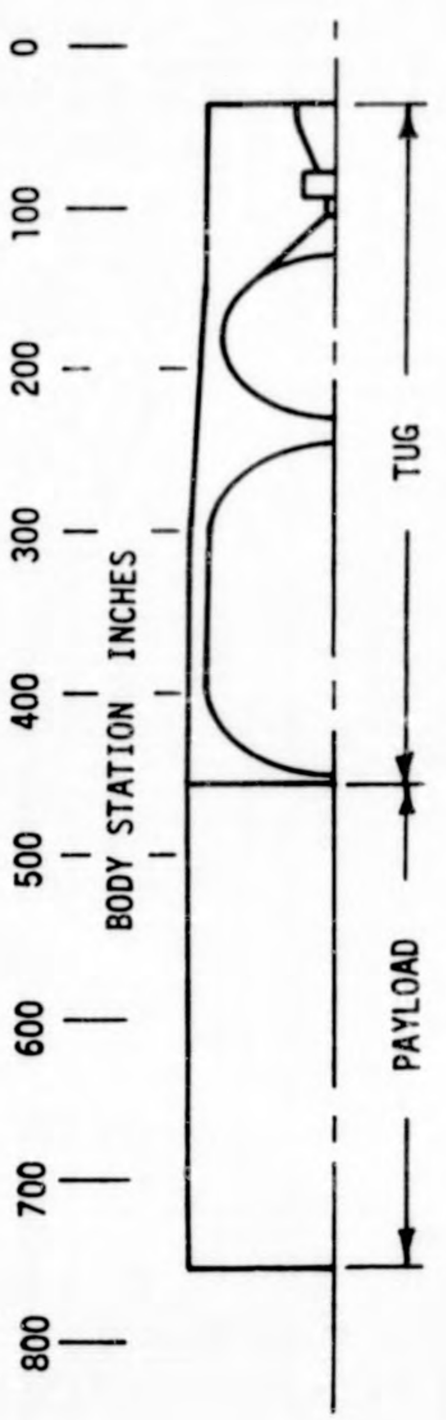
Table 8.3-1 Maximum Tug/Payload Body Deflections

ITEM	DEFLECTION (IN)	LOCATION
TUG LATERAL	1.6	STATION 304.5
TUG NORMAL	1.2	STATION 200
PAYLOAD LATERAL	0.6	STATION 452
PAYLOAD NORMAL	0.14	STATION 452



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North American Rockwell

8.3.3 Outer Shell Structural Analysis

Shell

A high-speed digital computer program (CØMHC) was used to define the preliminary design of the Tug shell. This program develops the dimensional configuration of cylindrical sandwich structures using rigorous analytical procedures for general stability, strength, core crimping and intercell skin crippling. A preliminary design is selected within the computer program from designs for several input orthotropic skin laminates, using minimum weight as the selection criterion. Program logic is based on the following:

(1) The selection of one of several input orthotropic skin materials is made on the basis of the minimum skin weight required to withstand applied loads. Minimum gage is used as a design restraint. Subsequent iterations for design selection consider different skin material. The total shell, sandwich, adhesive closeout, and frame weight from one material is compared to the weight of another material design, and a single, minimum weight design is selected for printout.

(2) The honeycomb depth is established using general stability equations developed for honeycomb sandwich shells. In addition to the longitudinal, circumferential and shear flexural rigidities, these equations include the shear rigidity of the core - a low shear rigidity resulting in core crimping and an empirical factor.

(3) Intercell skin crippling uses equations developed for conventional isotropic materials. These equations establish the maximum cell size.

Program input consists of skin orthotropic properties and minimum gage, honeycomb core density and physical properties, closeout materials, honeycomb bonding adhesive density, frame spacing and shell dimensions. Design restraints include maximum and minimum sandwich height.

Laminate properties used for input to the preliminary design program were calculated from the unidirectional properties defined in the materials section. For the point design of this study, the properties of the laminate were defined by the rule of mixtures and compared to laminate data from 1972 materials. For final design these calculated properties would be verified by more rigorous computerized laminate analysis techniques which account for residual stress and by test. The rule of mixtures, however, has been found to give reasonably accurate properties for the zero, 90 and 45 degree family of laminates used for the skins of the Tug shell. The properties used for each component at the point design are listed in Table 8.3-2.

Table 8.3-3 shows the calculated tensile and compressive load carrying capabilities of the shell components. These designs were based on the use of 2.2 lb/ft^3 aluminum honeycomb core and a minimum gage graphite epoxy laminate equal to 8 mil per skin. The load capabilities of each component may be compared with the body loads developed from axial forces and bending moments (see internal loads computations).

Table 8.3-2. Space Tug Skin Laminate Properties

Orientation[2]	Thickness (In.)	Tensile Strength Longi-tudinal	Hoop (psi)	Shear Strength (psi)	Poisson's Ratio	Longi-tudinal	Hoop	Shear	Remarks
FORWARD SKIRT									
0/+45/-45/90	0.008	54,000	54,000	28,000	0.31	9.2	9.2	3.5	Quasi-Isotropic Laminate
INTERTANK SHELL									
0/+45/-45/90	0.008	54,000	54,000	28,000	0.31	9.2	9.2	3.5	Quasi-Isotropic Laminate
AFT SKIRT									
0_2/ 45/-45/90	0.010	69,000	34,500	24,000	0.30	12.4	7.7	3.0	Quasi-Isotropic Plus One Extra Longitudinal Layer

NOTE: (1) Properties based on 1976 technology
(2) Orientation in degrees from longitudinal direction
(3) High tensile strength graphite epoxy laminate (density = 0.056 $lb/in.^3$)

Table 8.3-3. Shell Structural Capability

Component	Tensile Strength (lb/in.)	Compressive Crippling Load (lb/in.)
Forward Skirt	864	252
Intertank Shell	864	479
Aft Skirt	1389	1296

Tensile strengths of the forward skirt and intertank shell result from the minimum thickness restraint.

Edge closeouts were not sized by rigorous discontinuity or bolt bearing analyses for the point design. The magnitude of the maximum shell loads (1500 lb/in.), however, indicates that composite closeout doublers can be used for bolt bearing efficiency and that metal doublers are not required. The use of composite doublers for mechanical attachment at component edges minimizes the possibility of warpage resulting from the use of materials with dissimilar thermal coefficients. Isotropic laminates were selected for these doublers because this orientation provides good mechanical attachment performance considering the numerous modes of bolt bearing failure than can occur in composites - bearing, shear out, direct tension or hoop tension.

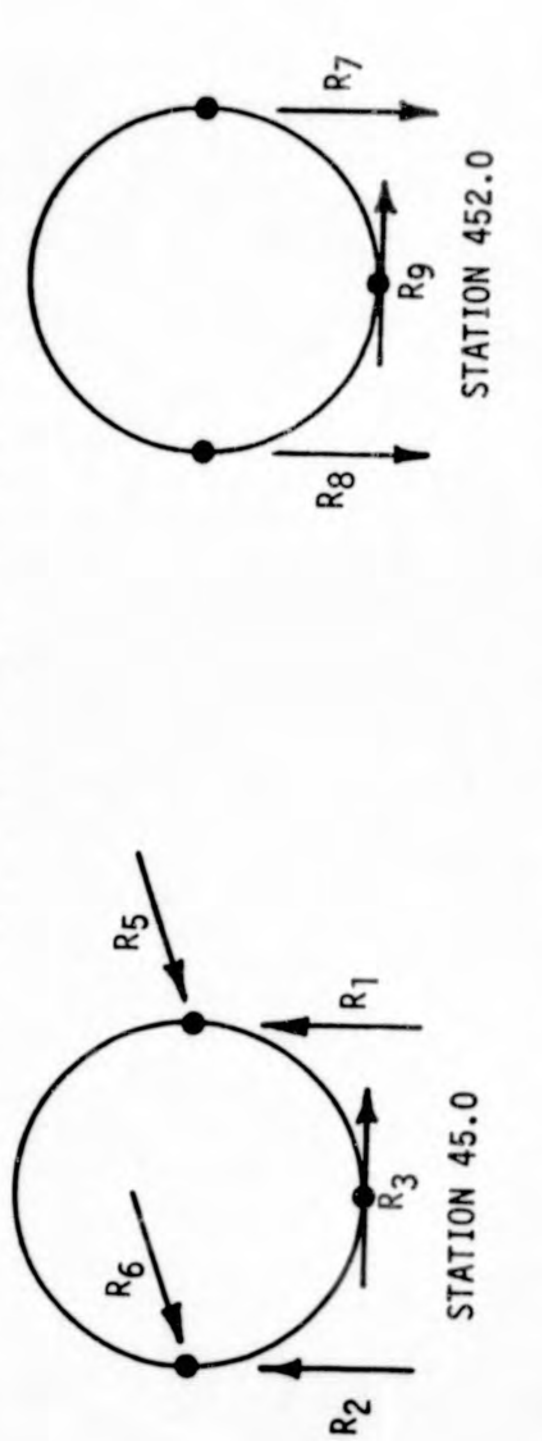
Stability frames were sized using Shanley's equation which relates frame flexural rigidity to spacing and axial load. The calculated frame rigidity from this equation is used to establish an area and weight on the basis of a preselected form factor. Frame spacing for the shell design was selected from the sandwich shell and frame weights using total minimum weight as the criterion. The flexural modulus of the frame assumed 80 percent circumferential graphite epoxy material in the caps.

The forward frame of the Tug (Station 452) is critical during orbiter launch. This condition imposes ultimate loads of 50 kip tangentially on two of three Shuttle cargo bay attach fittings. This frame is deflection critical. For the preliminary design, the critical tangential deflection, as calculated by the finite element internal loads analysis, was set at one inch. As for the stability frames, the caps of this frame were assumed to contain 80 percent circumferential graphite epoxy. Loads on this frame are shown in Table 8.3-4.

The critical loading for the mating frame at Station 126 occurs during launch. This condition imposes a maximum shell load of 900 lb/in. on the structure. For the preliminary design, a one-inch latch to shell neutral axis eccentricity was assumed and the resulting frame kick load calculated. The frame size was established by limiting radial deflections to 60 mil. The material orientation was the same as for the previously discussed frames.

Tank support frames were sized on the basis of local radial components of tank strut load. Torsional loads on the frames were minimized by locating

Table 8.3-4 Cargo Bay Attach Fitting Loads (Ultimate)



CONDITION \ REACTION	R_1	R_2	R_3	R_5	R_6	R_7	R_8	R_9
LAUNCH (FUEL PLUS PAYLOAD)	± 41,300 0	0 ± 41,300	± 41,300	+136,500	+136,500	± 49,700 0	0 ±49,700	±49,700
LAUNCH (FUEL, NO PAYLOAD)	± 50,316 0	0 ± 50,316	± 50,316	+130,200	+130,200	± 36,484 0	0 ±36,484	±36,484
END BOOST	± 24,780 0	0 ± 24,780	± 24,780	+150,150	+150,150	± 29,820 0	0 ±29,820	±29,820
MAX G	± 41,300 0	0 ± 41,300	± 41,300	- 9100	- 9100	± 49,700 0	0 ±49,700	±49,700
ENTRY	+ 8148 0	0 + 8148	± 2038	- 3681	- 3681	+ 31,720 + 19,032	+19,032 +31,720	±12,688
LANDING	+ 6518 0	0 + 6518	± 1018	- 9572	- 9572	+ 23,473 + 17,129	+ 17,129 + 23,473	± 6344

intersection of shell neutral axis and strut loads on the centerline of the frames. Ultimate radial design loads on the LOX and LH_2 frames were 2250 lb and 640 lb per attach point, respectively. A factor of three was applied to these loads to account for other than uniform radial load conditions.

Adapter

The adapter skin thicknesses were established by iterative runs of a computerized shear lag program. This program, titled "Stress Analysis by Shear Lag Relaxation Procedure" (Program XG 0003), calculates deflections and stresses in a sheet-stringer structure (S-S-Struct.) subjected to concentrated mechanical and thermal loads applied along the stringers. Sandwich structures are analyzed by considering the longitudinal extensional rigidity of the skins to be grouped into an equivalent stringer system. The stringers and an effective part of the sheet connecting the stringers are considered to behave as axial members absorbing tension or compression loads. The sheet areas absorb any shear loads that may result from a combination of cutouts, non-uniform stiffness of structure, and some arbitrary loading of the structure.

The point design preliminary sizing was based on the maximum axial load condition of 150 KIP per attachment point (see end-boost condition of Table 8.3-4). Since the critical station for near uniform load distribution is at the LOX tank support frame (Station 156), the adapter and aft skirt were both included in the shear lag analysis of the load distribution. A model for the adapter and aft skirt was established (see Figure 8.3-12) and analyses made. The stiffness of the high shear area of the adapter skins was modified in successive runs until the average running load was greater than two-thirds the peak load.

Calculated running loads are shown in Figure 8.3-12. Based on this distribution and the thicknesses of the shell, the maximum stresses in the aft strut and adapter will be 27 ksi and 44 ksi, respectively. The maximum shear stress in the adapter is 9700 psi compared to an ultimate of 18,000 psi. The low magnitude of the applied shear stress indicates that the adapter is stiffness critical rather than strength critical.

For final adapter design, the shear lag program used in this study would be the first analytical step. After preliminary sizing by this computer program, a more detailed analysis would be made using a finite element program such as NARSAMS or ASKA. The internal loads analysis of the total shell used NARSAMS. Running loads from this program were comparable to the shear lag results.

The adapter uses two major frames - at the forward and aft ends. The frame at the forward end (Station 126) is identical to the aft skirt frame at the same station. It is discussed in the aft skirt section. The other frame (Station 45) reacts lateral attachment loads at the aft end of the adapter. This frame was designed by the stiffness required to limit tangential deflections to less than one inch. The stiffness analysis to verify compliance to this requirement was made as a part of the internal loads analysis. The critical design condition for the frame results in tangential ultimate loads of approximate 50 KIPS on two adjacent attachment fittings. The frame reaction loads are shown in Table 8.3-4.

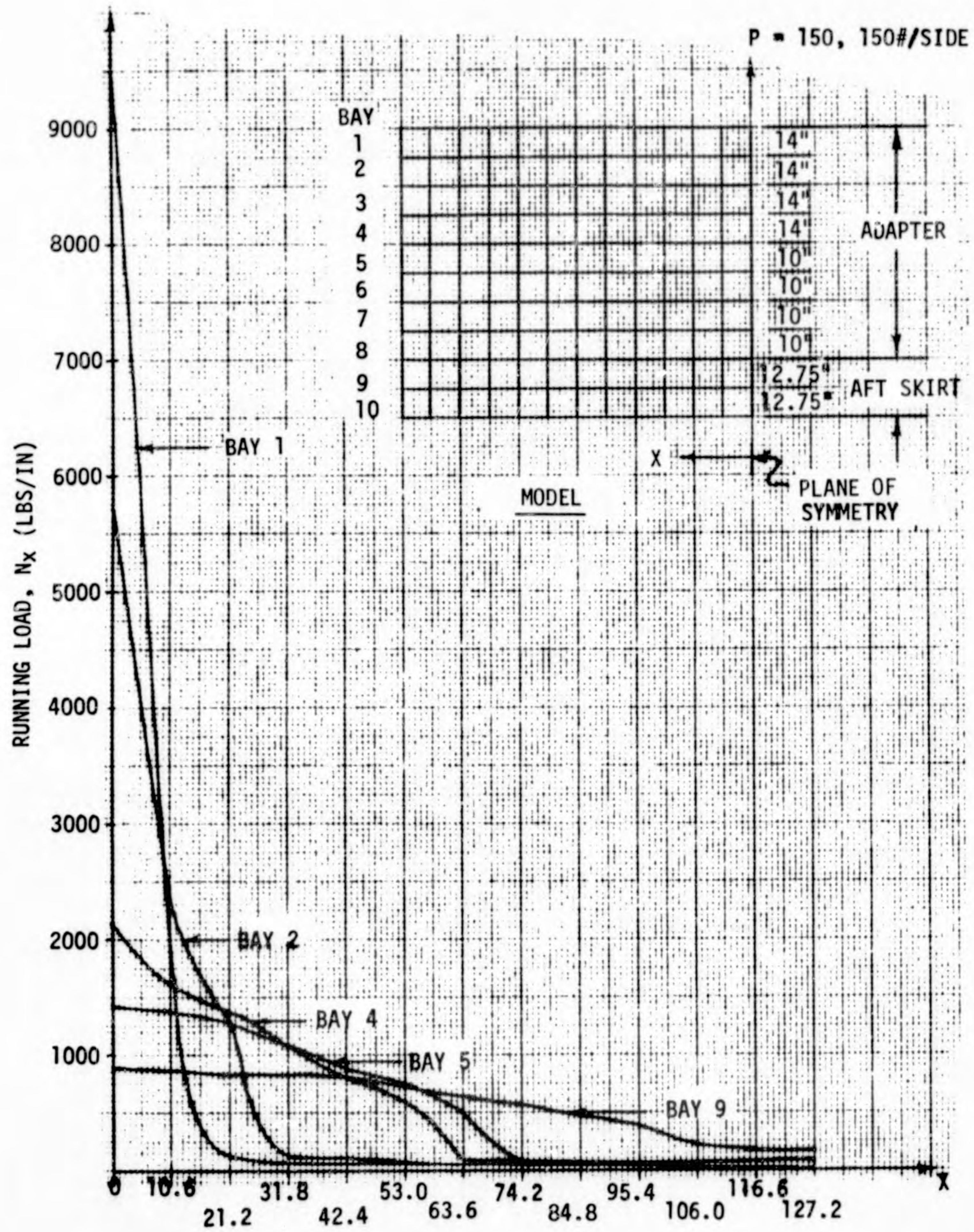


Figure 8.3-12 Load vs Distance From Attach Point

The adapter contains two intermediate frames. These frames were of greater cross section than the section required for shell stability. This increase in section was made to permit attachment of internal system hardware such as purge gas pressure bottles.

Thrust Structure

General Description

As shown in Figure 8.3-5, the thrust structure is a conical arrangement of frame and skin-stabilized tubular struts attached to the LOX tank aft bulkhead. Its primary function is to react symmetric and unsymmetric loads from engine thrust and distribute them to the bulkhead shell. In addition, the frames react external loads from engine gimbal actuators and from various systems components and engine feedlines attached to the structure. Torsional stability is provided by a conical skin attached externally to the struts and extending from the engine thrust block fitting to the forward frame. This skin also provides meteoroid protection and acts as the purge membrane for the aft portion of the thrust structure. Two additional skin shear panels extending from the forward frame to the LOX bulkhead attach fittings provide the required torsional stability for the forward bay of the thrust structure.

This design reduces to an acceptable level the thrust loads introduced to the tank shell. Bulkhead deflections and rotations at the strut attach points are accommodated by the flexibility of the forward bay struts without the introduction of high secondary loads. Heat leak to the LOX tank is minimized by material selection and the minimum conductive areas of the structural elements.

Construction

The 12 tubular struts are approximately 60 inches in length. They are fabricated from boron- and graphite-reinforced epoxy composites on a 1.50-inch diameter mandrel. The resulting cross section consists of inner and outer hoop layers of 0.005-inch HTS graphite epoxy. Between these is a 0.0105-inch thick layer of longitudinal fiber boron epoxy. Two parallel 0.60-inch flats on opposite sides provide surfaces for attachment to skin and frames. Glass cloth reinforced epoxy end fittings are bonded to the strut ends for mechanical attachment to the thrust block and bulkhead shell. These fittings effectively provide the column fixity at the thrust block required for thrust structure stability. The bulkhead attach fittings are contoured adjacent to the shell to minimize eccentricity.

Two channel section graphite epoxy ring frames provide general stability to the thrust structure by reducing the effective column lengths of the struts. Both are attached perpendicular to the conical contour for maximum efficiency. The aft frame is internal to the structure; height is 1.60 inches with cap width and thickness 0.40- and 0.08 inches respectively. The forward frame is attached externally to the conical structure due to proximity to the LOX bulkhead. Height is 1.80 inches; caps are 0.45 inch wide and 0.09 inch thick.

A 0.010 inch thick quasi-isotropic glass cloth reinforced epoxy skin extends from the thrust block attach fittings to the forward ring frame. It is attached by mechanical fasteners to the external flat surfaces of the thrust structure struts and to the inboard cap of the forward frame. This skin is sized to react purge system differential pressure, to provide meteoroid protection for the aft portion of the vehicle and for torsional stability of the conical structure. Torsional stability of the upper bay of the thrust structure is provided by two additional glass cloth reinforced epoxy skin shear panels symmetrically located between adjacent struts and extending from the forward frame to the LOX bulkhead attach fittings. As for the aft skin, attachment is made to the struts by mechanical fasteners.

Design Description

Engine thrust loads are reacted in axial compression by the 12 tubular struts. Straight ahead (zero gimbal) thrust produces equal load in each strut. The critical loading, however, results from the full seven degree gimbaled thrust condition which increases the maximum strut load approximately 17.6% above the zero gimbal value. This maximum load is used for strut design and is assumed to exist simultaneously in all struts for determination of stability frame requirements. Struts are designed to preclude both general and local stability failures under design ultimate loads.

Although attached to the thrust block, the forward frame, and to each strut between these members, the skin reacts less than 1 percent of the thrust load because of its low modulus. For purposes of strut design, the skin is considered to carry no thrust load and to be ineffective in the determination of strut crippling allowables. The skin is therefore sized to react differential pressures from internal purge gas and to meet meteoroid protection thickness requirements. The skin also provides torsional stability to the conical structure although specific requirements for torsional stiffners and stability have not as yet been established.

Frame bending stiffness requirements were determined to preclude general cone instability prior to general or local instability of the individual struts. To enable these frames to react external loads from engine gimbal actuators and various attached systems, section property requirements were determined to react a radial load of 100 pounds. This latter condition proved critical in sizing the two thrust structure frames.

Structural Analysis

Strut loads were calculated from the geometry of the structure and the applied thrust load using elementary statics. For the 12 strut, 53-degree apex half-angle conical structure the strut loads are as follows:

$$P = 0.1385 \times \text{Thrust } (0° \text{ gimbal})$$

$$P = 0.1629 \times \text{Thrust } (7° \text{ gimbal})$$

For a limit thrust of 10,000 pounds and ultimate factor of safety of 1.25, the maximum design strut load is 2040 pounds compression with 7° gimbal angle. The only tension loading on the struts results from shuttle re-entry and is of negligible magnitude.

Struts were analyzed for general and local stability. General stability allowables were determined for the longest span of approximately 27-inches between the two frames using the classical Euler equations and assuming ends pinned. Local crippling allowables were determined for the circular contour using the Dow and Rosen theory of Reference 3 and for the flats using techniques of Reference 5.

Frame bending stiffness (EI) requirements necessary to provide general stability were determined by using the Shanley equation for cylindrical stiffened shells in bending

$$EI_{reqd} = \frac{MR^2}{4000L}.$$

Thrust structure loading was idealized to accommodate this equation by assuming the maximum strut load to result entirely from applied bending moments (conservative). This load was then converted to an equivalent uniform loading. The resulting equation is

$$EI_{reqd} = 3.16 \frac{R^3}{L}.$$

For additional conservatism, the radius R was taken as being perpendicular to the struts at the frame. Frame moment of inertia values were calculated from the minimum bending stiffnesses thus determined. These were found to be insufficient to react a 100 pound radial load representing typical attached lines and systems. The two frames were thus sized to meet this arbitrary condition using the Peenumunde coefficients for internal loads.

The conical skin was sized to provide the required meteoroid protection. The thickness thus determined exceeded that required to react the purge gas design ultimate differential pressure in membrane tension.

Purge and Meteoroid Protection Closeouts

Two purge and meteoroid protection (P and M) closeouts are employed on the Space Tug. These items are so classified because they serve dual functions of containing purge gas within the outer shell and providing meteoroid protection for portions of the vehicle. No primary structural loads are reacted by these items.

As shown in Figure 8.3-3, the forward P and M closeout is a flexible membrane attached near the inboard cap of the forward skirt X_B 420.5 frame. Preformed to an approximately 140-inch spherical radius of curvature, it reacts purge gas differential pressures and provides meteoroid protection for the forward LH_2 bulkhead. One opening through the membrane is required to

accommodate the LH_2 fill and drain line. A seal is provided around this penetration to prevent leakage of purge gas and provide the required meteoroid protection.

The membrane is constructed of rubber impregnated, 0.016 inch thick, quasi-isotropic, type 112 glass cloth. Since it has little bending stiffness, internal purge pressure is reacted in pure membrane tension in the preformed spherical contour. Continuous peripheral support is provided by mechanical fasteners attached through a ring which clamps the membrane to the frame.

The forward P and M closeout is sized primarily to provide the required meteoroid protection as discussed in section 7.0. Loads from the 0.28 psi design ultimate purge pressure differential were determined using classical membrane theory. These were found to produce negligible stresses in the 0.016- inch thick glass cloth.

The aft P and M closeout, as shown on Figure 8.3-5, is a rigid shell in the form of a truncated cone. Of honeycomb construction, it is supported at the larger end by attachment to the inboard cap of the aft skirt X_B 156 frame. The cone extends aft to the smaller end which is connected to the inboard cap of the forward thrust structure frame by a flexible seal. It reacts purge gas differential pressures and provides meteoroid protection over the area between the thrust structure skin and the aft skirt.

The cone is constructed of 0.008-inch thick quasi-isotropic graphic epoxy facing sheets separated by 0.25-inch thick, 1/4-inch cell size, aluminum core. A graphite epoxy closeout around the larger periphery incorporates an extension for mechanical attachment to the frame. The closeout at the smaller end enables mechanical attachment to the flexible seal.

This design incorporates the rigidity required to maintain the conical shape under design loadings. The flexible seal connection to the thrust structure frame reacts the purge gas differential pressure while avoiding a redundant load path between the thrust structure and the aft skirt. The sandwich construction provides capability to react unsymmetric loadings from lateral inertia forces.

The aft P and M closeout is analyzed as an internally pressurized conical shell, simply supported about the larger periphery using methods of Reference (4). The cone was conservatively assumed to act as a simple shell with thickness equal to that of the two sandwich facing sheets. A second independent calculation was performed to verify the ability of a unit strip, parallel to an element of the cone, to react the purge pressure in bending assuming simple end supports. Both calculations demonstrated the structural integrity under the 0.28 psi design ultimate purge pressure differential.

8.4 PRESSURIZED STRUCTURE AND SUPPORT SUBSYSTEM

This section presents the structural analysis of the main propellant tankage and support system.

The main propellant tankage consists of a LH_2 tank and a LOX tank. These tanks are "floating" type tanks which are supported within the outer shell by a fiberglass composite truss. The LH_2 tank is a short cylindrical section closed out by two hemispherical bulkheads. The LOX tank is an ellipsoid of revolution with an aspect ratio of 1.43 to 1.

The methods of analysis used in this section are conventional in general and where lengthy derivations or computer applications are required, separate reports are referenced.

8.4.1 Structural Description

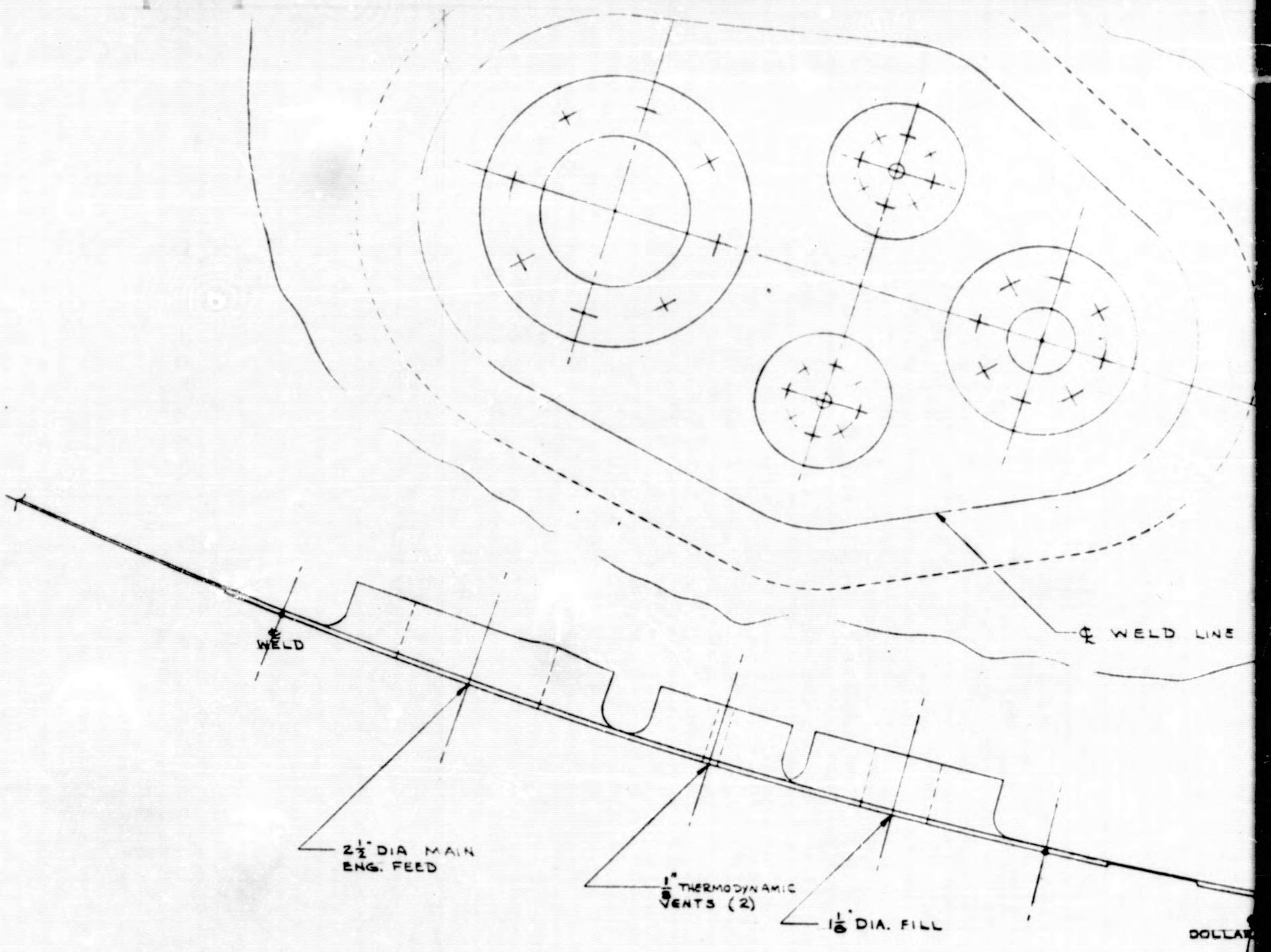
LH_2 Tank and Support System

The main LH_2 tank as shown on Figure 8.4-1 is a cylindrical, 2014-T651 aluminum alloy tank 168 inches in diameter with convex, hemispherical bulkheads at each end. It volume is 1916 cubic feet with a capacity of 8105 pounds of LH_2.

The end bulkheads consist of six preformed gores and a central circular section, all butt welded together. The main cylindrical section is made up of 3 sheets (.10 inch thick) butt welded together to form a cylinder. This cylinder is chem-milled to a thickness of .045 inches between the weld lands. A heavier ring segment is welded between the aft bulkhead and the main cylindrical section. This ring provides the thickness necessary to attach the strut support fittings. This heavy section also serves to distribute the loads introduced by the tank support struts. The end bulkheads are, essentially, .080 inches thick sheet chem-milled to .020 inches between the weld lands. In certain areas, heavier bosses are welded in to provide attach points for propellant feed lines, etc. The forward bulkhead has an access door located in the center of the circular section. The aft bulkhead has an identical size circular section but with structural bosses and no access door. The ACPS start tank is mounted inside the main LH_2 tank. It is located 3 inches off the tank center line and is supported one inch above the bottom of the tank on 6 pairs of lugs welded 1-3/8 inches apart on the inside of the gore weld lands. The LH_2 tank assembly is supported within the TUG shell structure by a series of 48 "S" glass filament wound composite, tubular struts. The struts are attached to the LH_2 tank by fittings bolted to the heavy ring section at 15° intervals.

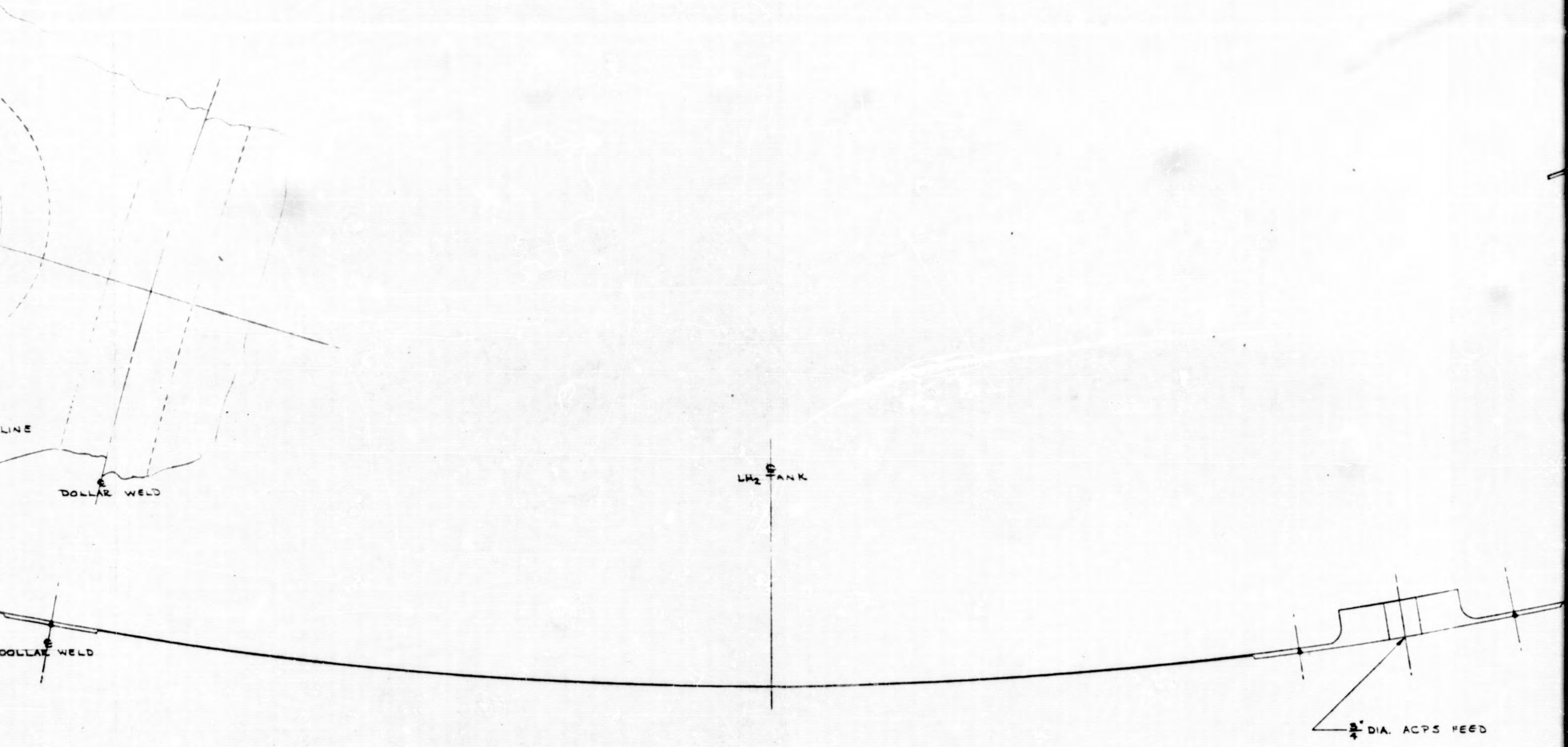
LOX Tank and Support System

The LOX tank shown on Figure 8.4-2 is an ellipsoidal shaped structure consisting of welded 2014-T651 aluminum alloy bulkheads. It is supported within the Tug shell structure by 48 "S" glass filament wound composite tubular struts. The volume is 712 cubic feet with a capacity of 48,633 lbs of LOX. Each bulkhead is manufactured from 6 preformed gores butt welded



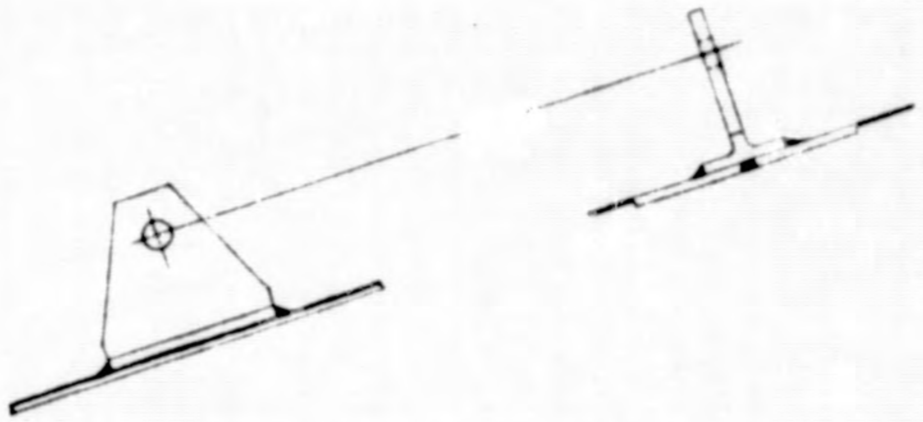
WELD
$2\frac{1}{2}$" DIA MAIN ENG. FEED
$\frac{1}{8}$" THERMODYNAMIC VENTS (2)
$1\frac{1}{8}$" DIA. FILL
℄ WELD LINE
DOLLAR

F-1

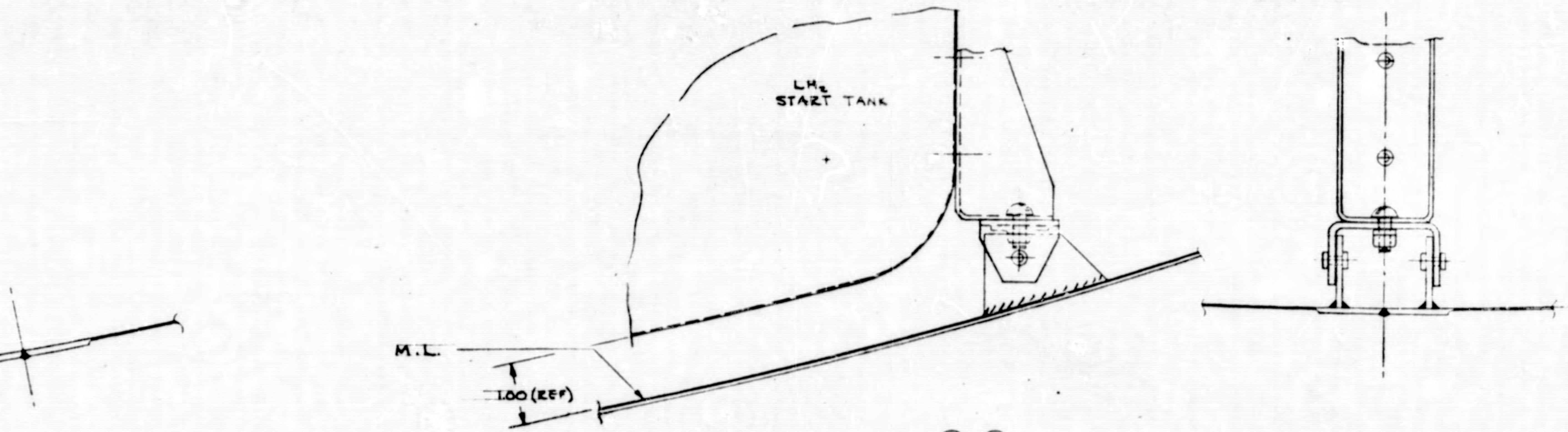


SECTION R-R
FULL SIZE

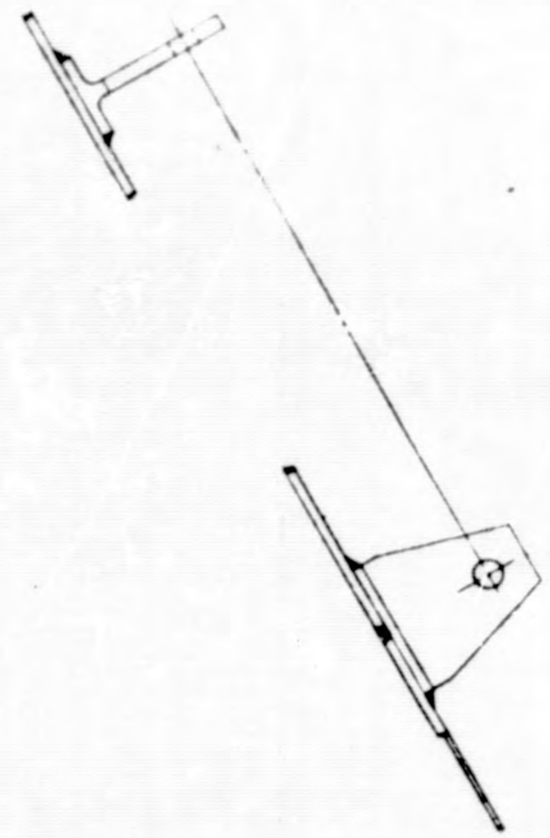
F-2



SECTION U-U
FULL SIZE
BAFFLE SUPPORT LUG
TYP 12 PLACES

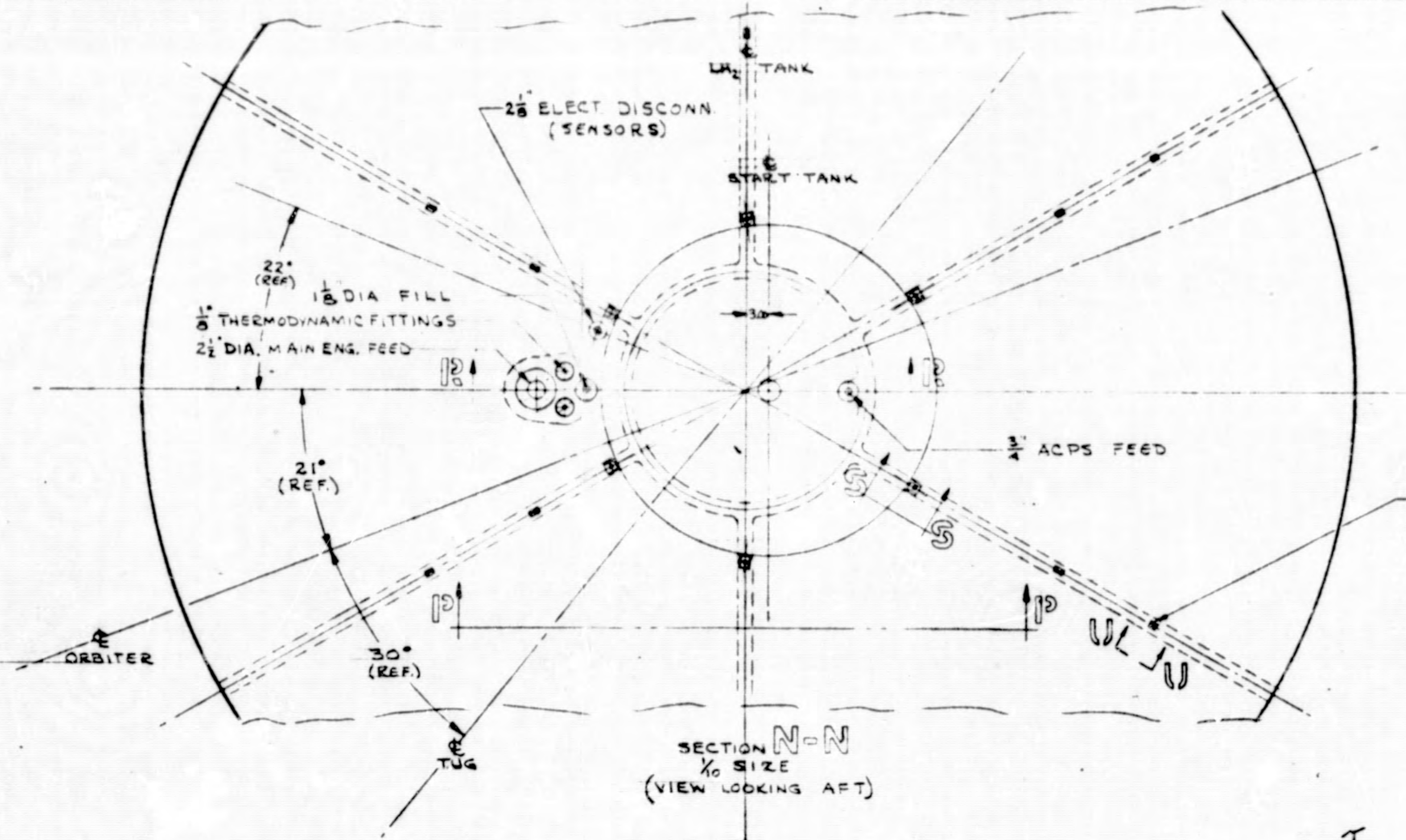


SECTION S-S
FULL SIZE
(TYPICAL SIX PLACES)

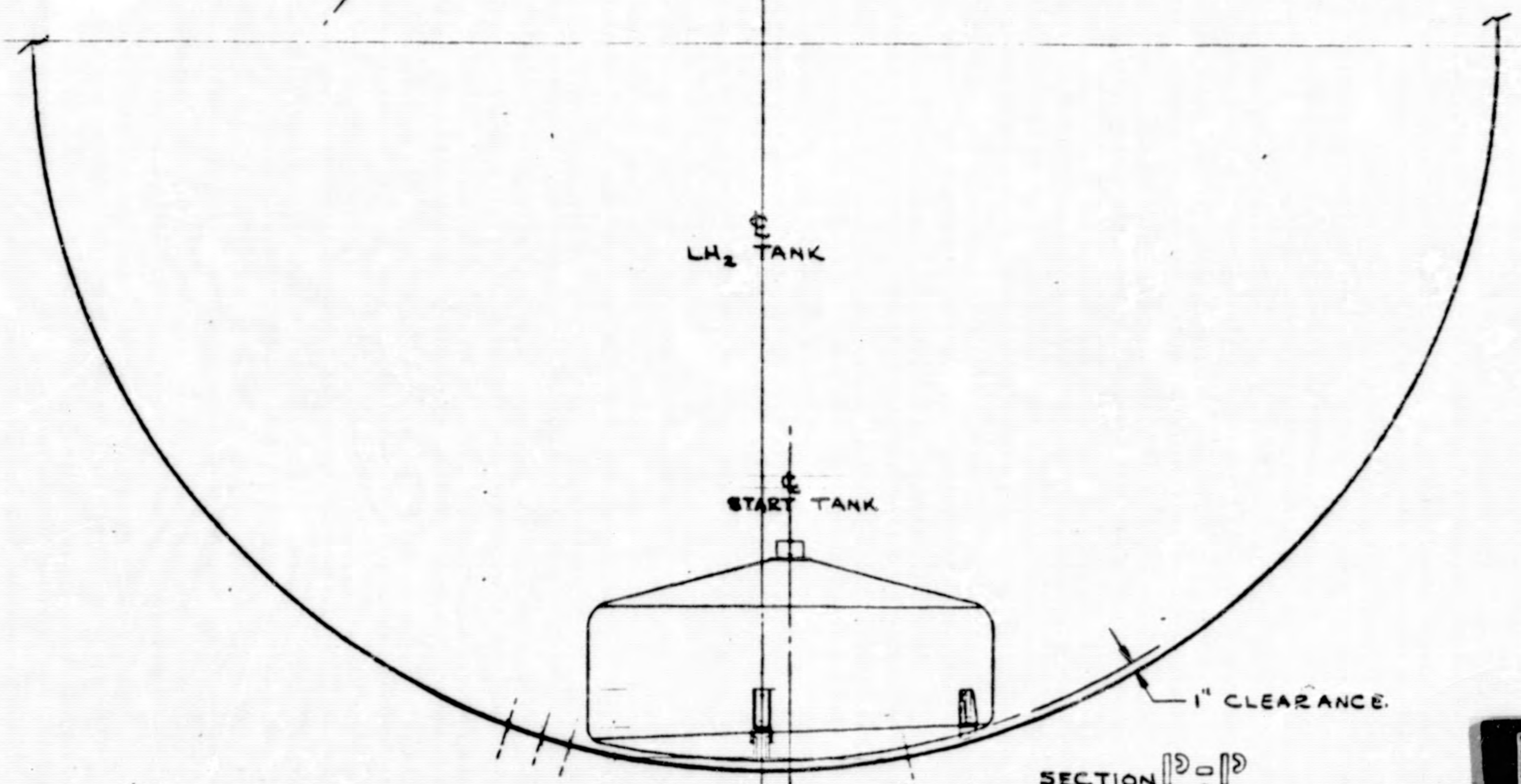


SECTION T-T
FULL SIZE
POINT SENSOR MAST GUY ATTACH LUG
3 LUGS LOCATED AT JUNCTION OF FWD BHD & CYLINDRICAL SECTION
3 LUGS LOCATED AT JUNCTION OF AFT BHD & CYLINDRICAL SECTION
AT 126° 30', 215° 30', & 351°

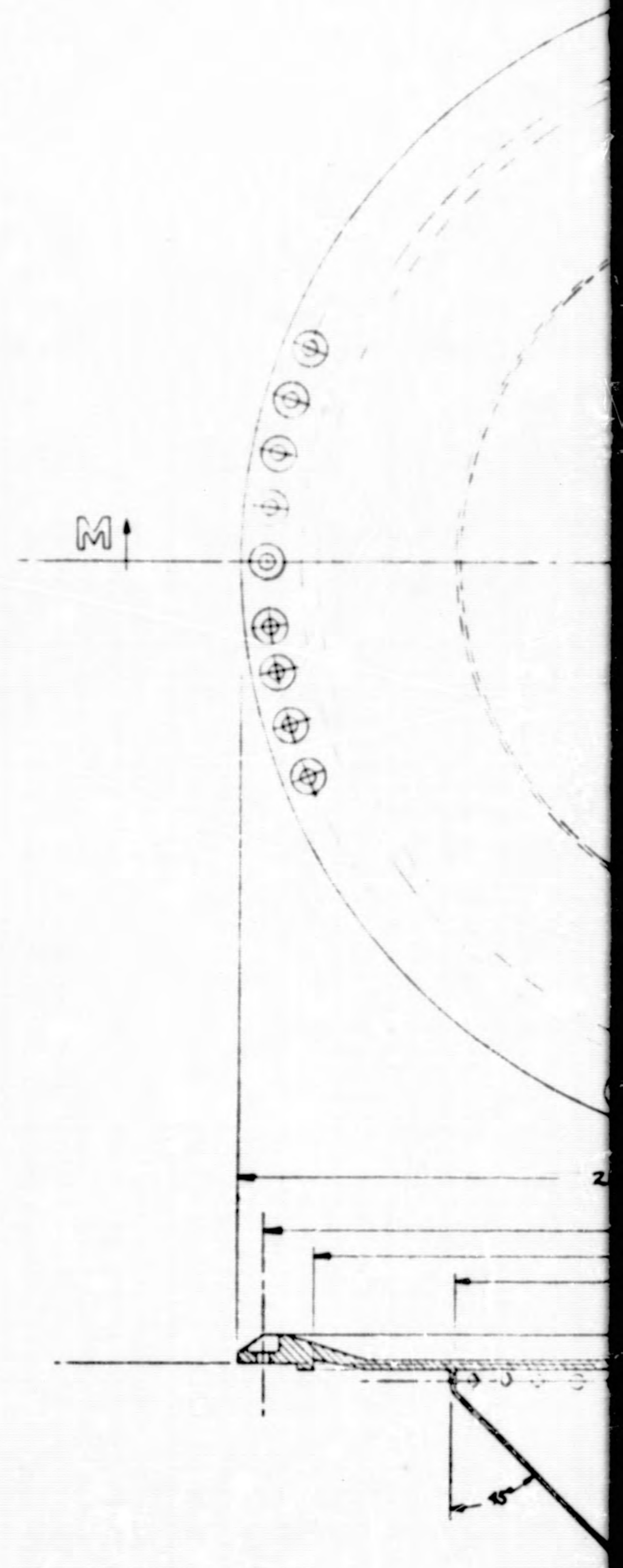
F-3



SECTION N-N
1/10 SIZE
(VIEW LOOKING AFT)

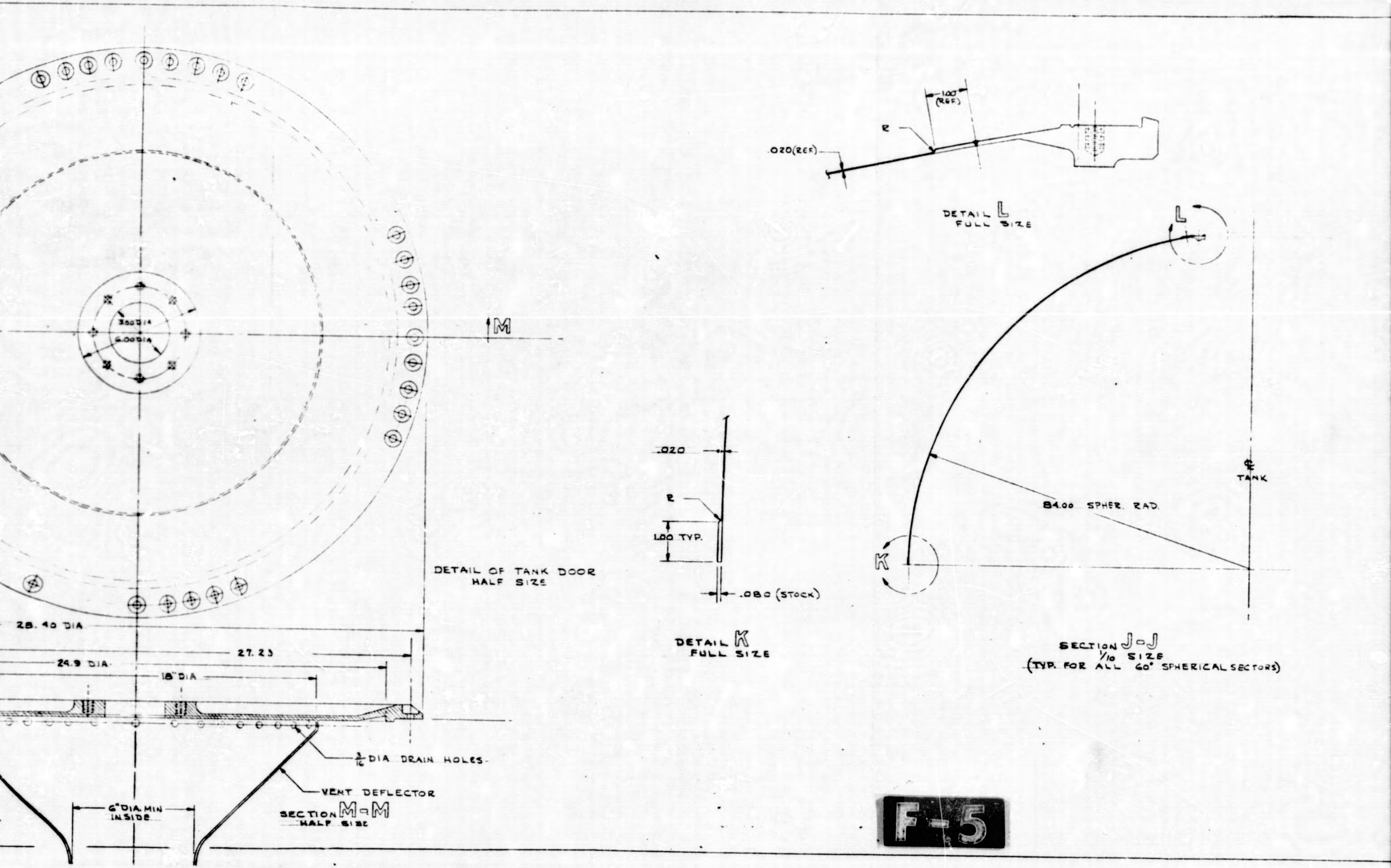


SECTION P-P
1/10 SIZE



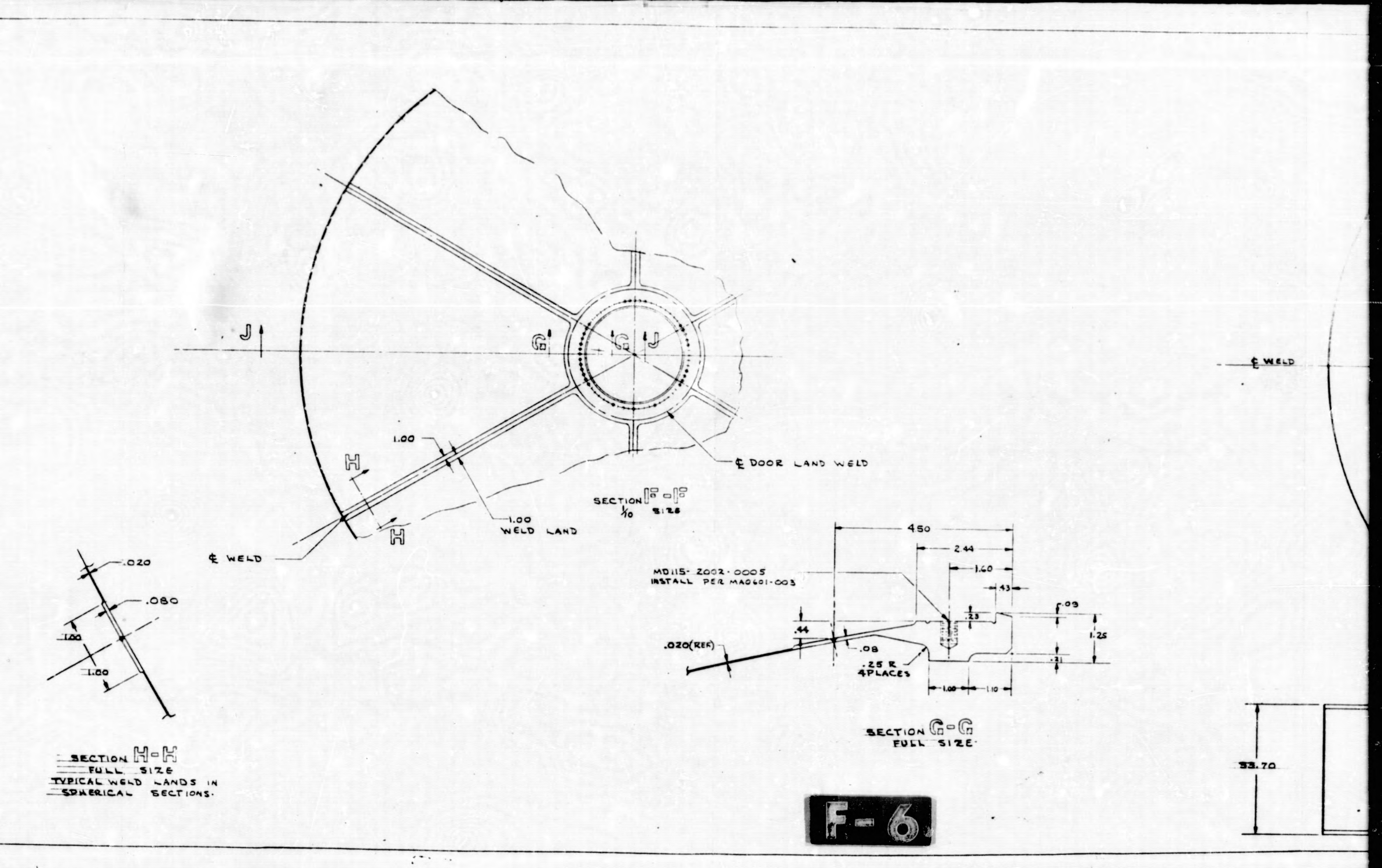
ICAL SECTION
RICAL SECTION

F-4

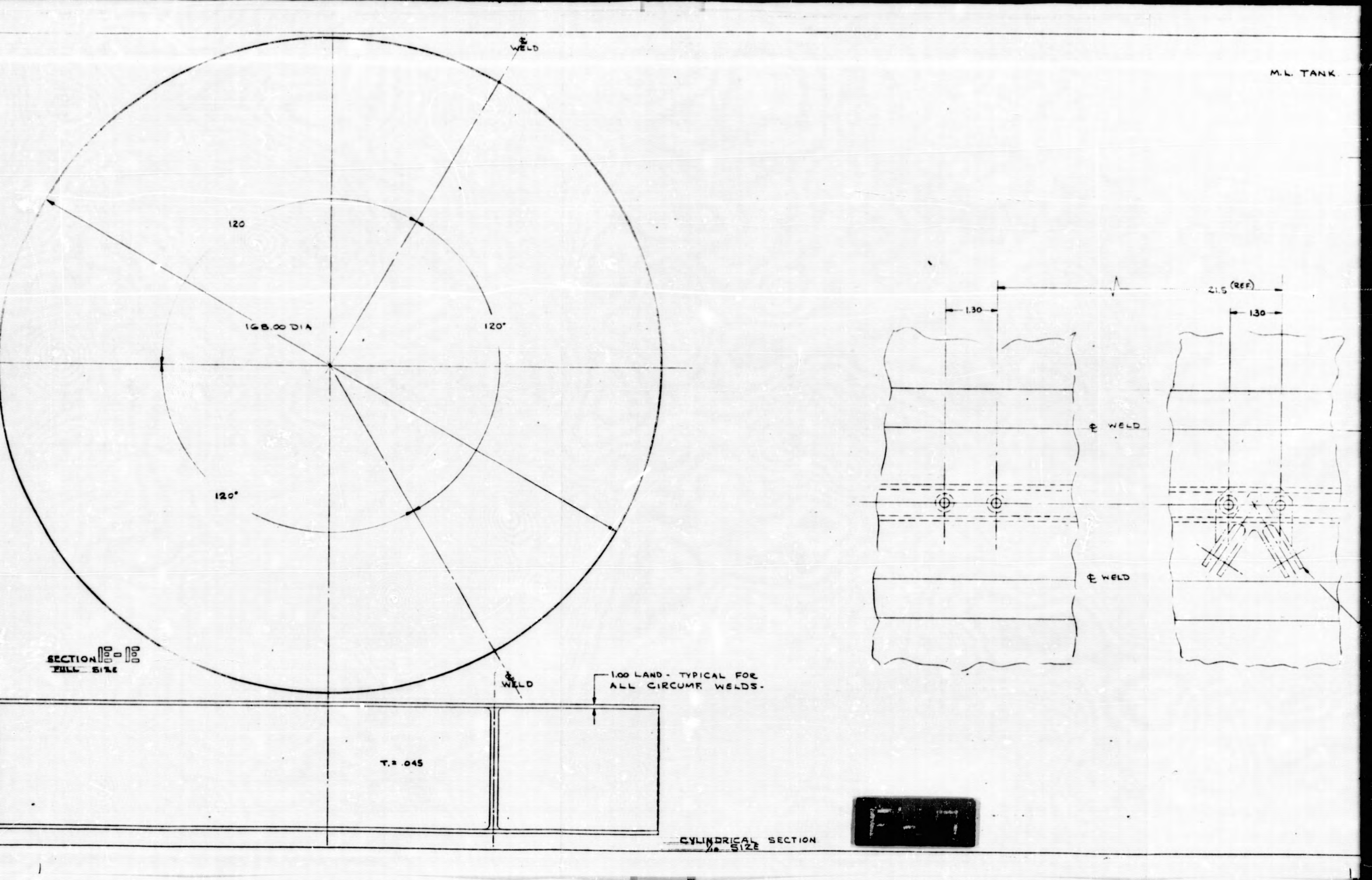


3.00 DIA
6.00 DIA
M
DETAIL OF TANK DOOR
HALF SIZE
28.40 DIA
27.23
24.9 DIA
18" DIA
3/16 DIA DRAIN HOLES
VENT DEFLECTOR
6" DIA MIN
INSIDE
SECTION M-M
HALF SIZE
.100
(REF)
R
.020 (REF)
DETAIL L
FULL SIZE
.020
R
1.00 TYP.
.080 (STOCK)
DETAIL K
FULL SIZE
L
K
84.00 SPHER. RAD.
℄
TANK
SECTION J-J
1/10 SIZE
(TYP. FOR ALL 60° SPHERICAL SECTORS)

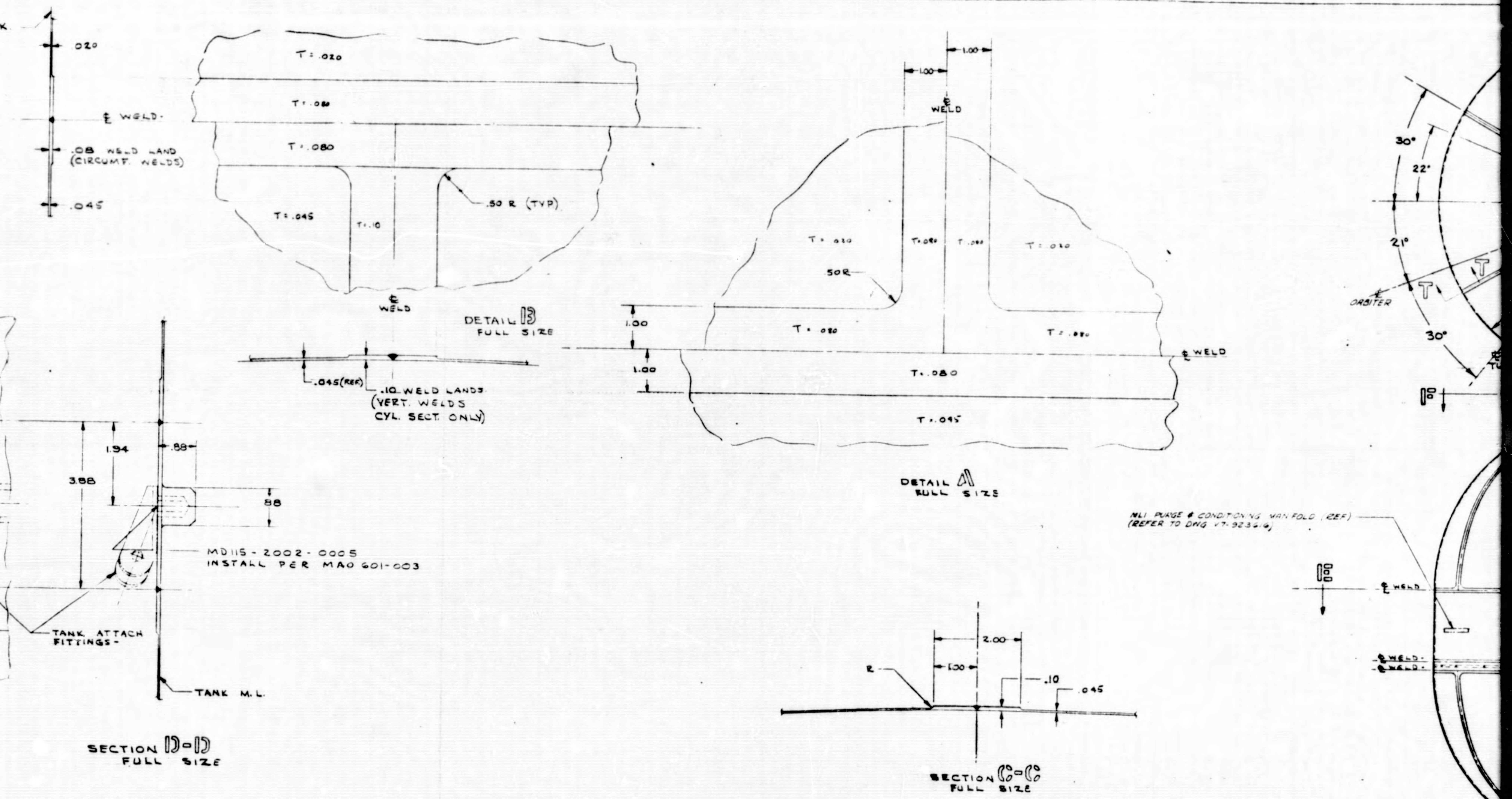
F-5


J
G
G J
℄ WELD
1.00
H
H
℄ DOOR LAND WELD
SECTION F-F
1/10 SIZE
1.00
WELD LAND
℄ WELD
.020
.080
1.00
1.00
4.50
2.44
1.60
.43
MD115-2002-0005
INSTALL PER MA0601-003
.09
.23
1.25
.44
.08
.020(REF)
.25 R
4 PLACES
.21
1.00
1.10
SECTION G-G
FULL SIZE
SECTION H-H
FULL SIZE
TYPICAL WELD LANDS IN
SPHERICAL SECTIONS.
53.70

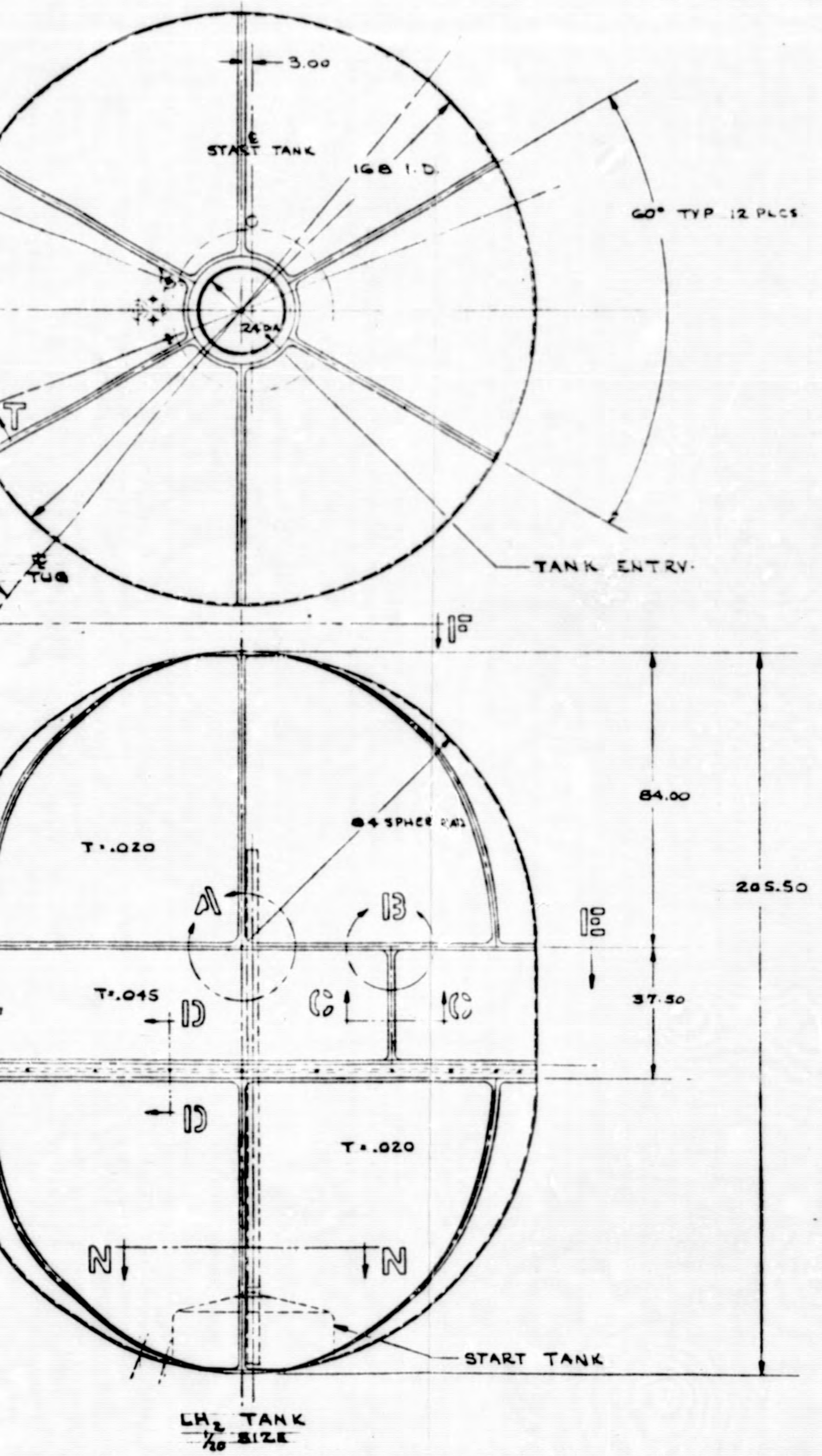
F-6



M.L. TANK
℄ WELD
120
168.00 DIA
120°
120°
SECTION E-E
FULL SIZE
℄ WELD
1.00 LAND - TYPICAL FOR ALL CIRCUMF. WELDS-
T. = .045
CYLINDRICAL SECTION
1/10 SIZE
1.30
21.5 (REF)
1.30
℄ WELD
℄ WELD


.020
℄ WELD
.08 WELD LAND
(CIRCUMF. WELDS)
.045
T=.020
T=.080
T=.080
T=.045
T=.10
.50 R (TYP)
℄ WELD
DETAIL B
FULL SIZE
.045(REF)
.10 WELD LANDS
(VERT. WELDS
CYL. SECT. ONLY)
1.00
1.00
℄ WELD
T=.020
T=.080
T=.080
T=.020
.50R
1.00
T=.080
T=.080
℄ WELD
1.00
T=.080
T=.045
DETAIL A
FULL SIZE
30°
22°
21°
ORBITER
30°
1.94
.88
3.88
.88
MD115-2002-0005
INSTALL PER MAO 601-003
TANK ATTACH
FITTINGS
TANK M.L.
SECTION D-D
FULL SIZE
MLI PURGE & CONDITIONING MANIFOLD (REF)
(REFER TO DWG VT-923616)
℄ WELD
℄ WELD
℄ WELD
2.00
1.00
R
.10
.045
SECTION C-C
FULL SIZE

[redacted]



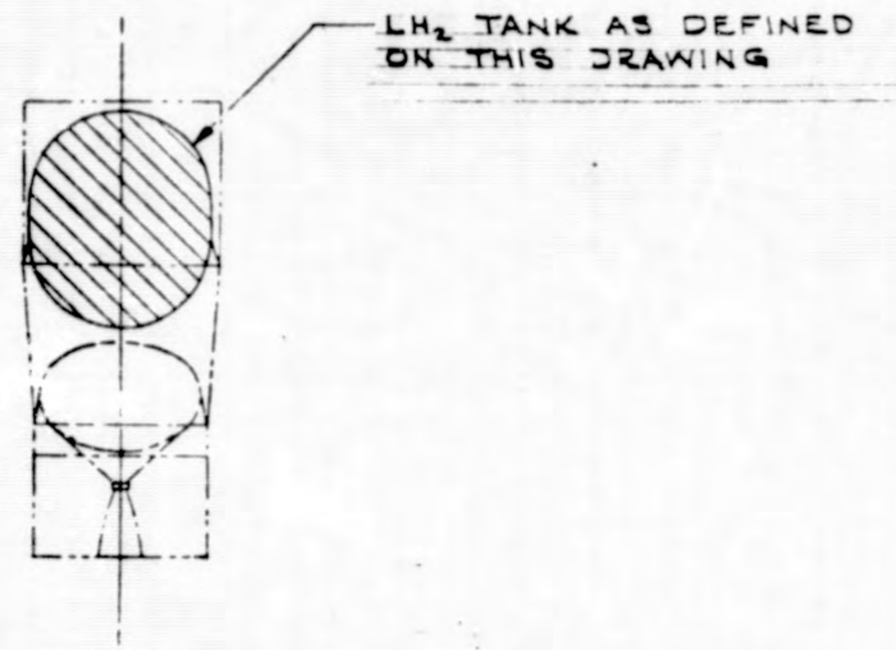


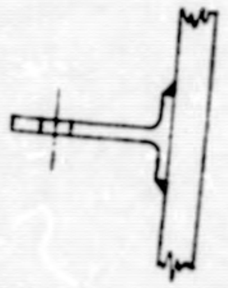
Figure 8.4-1 LH_2 Tank Tug

INFORMATION ONLY

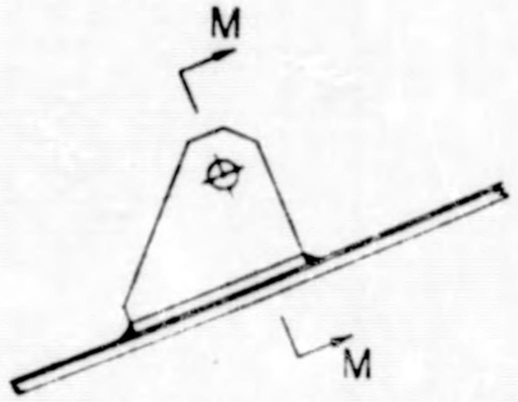
2. FOR PROPELLANT ORIENTATION & MANAGEMENT SYSTEMS INSTALLATION REFER TO DWG V7-949111
1. TANK MATERIAL: 2014-T6 AL. ALLOY SHEET (MBO-170-002)

NOTES:

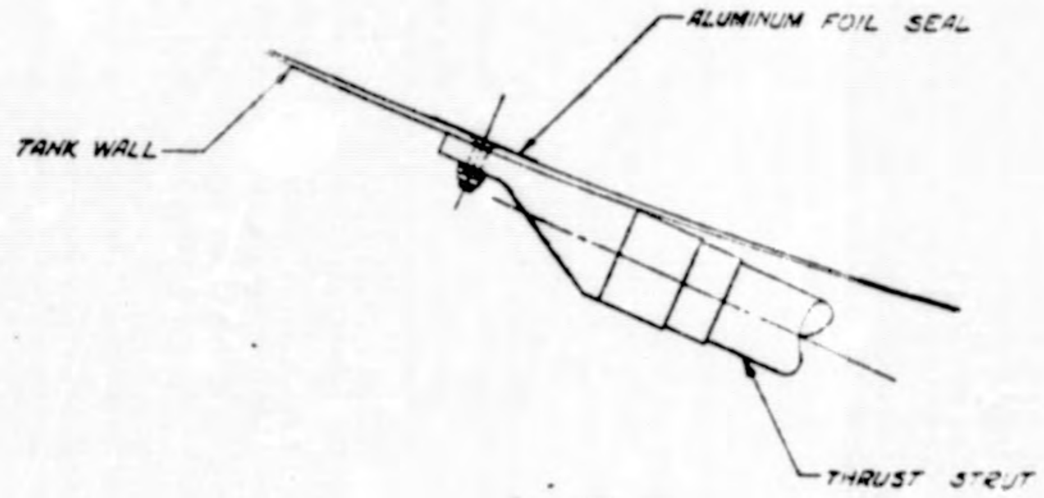
SCALE NOTED	DR. SWENSON / DATE 12 DEC 71 / CHECK	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	LH_2 TANK TUG	V7-923608



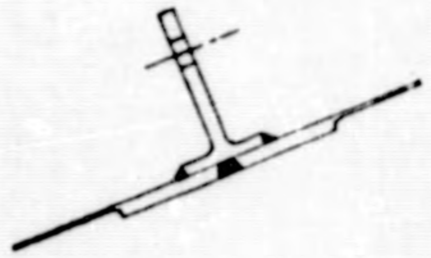
SECTION L-L
SCALE 1/1



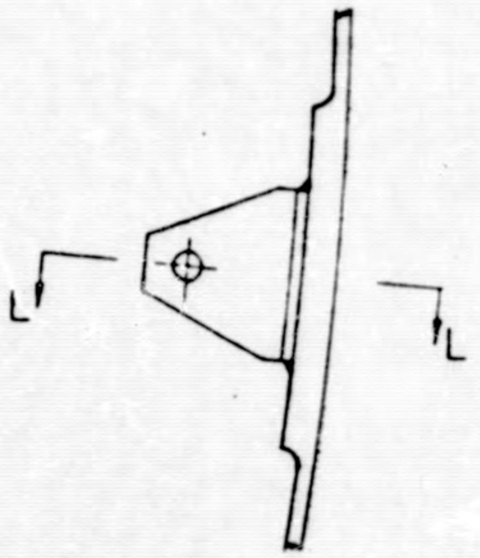
SECTION J-J
SCALE 1/1
BAFFLE SUPPORT LUG
TYP 12 PLACES



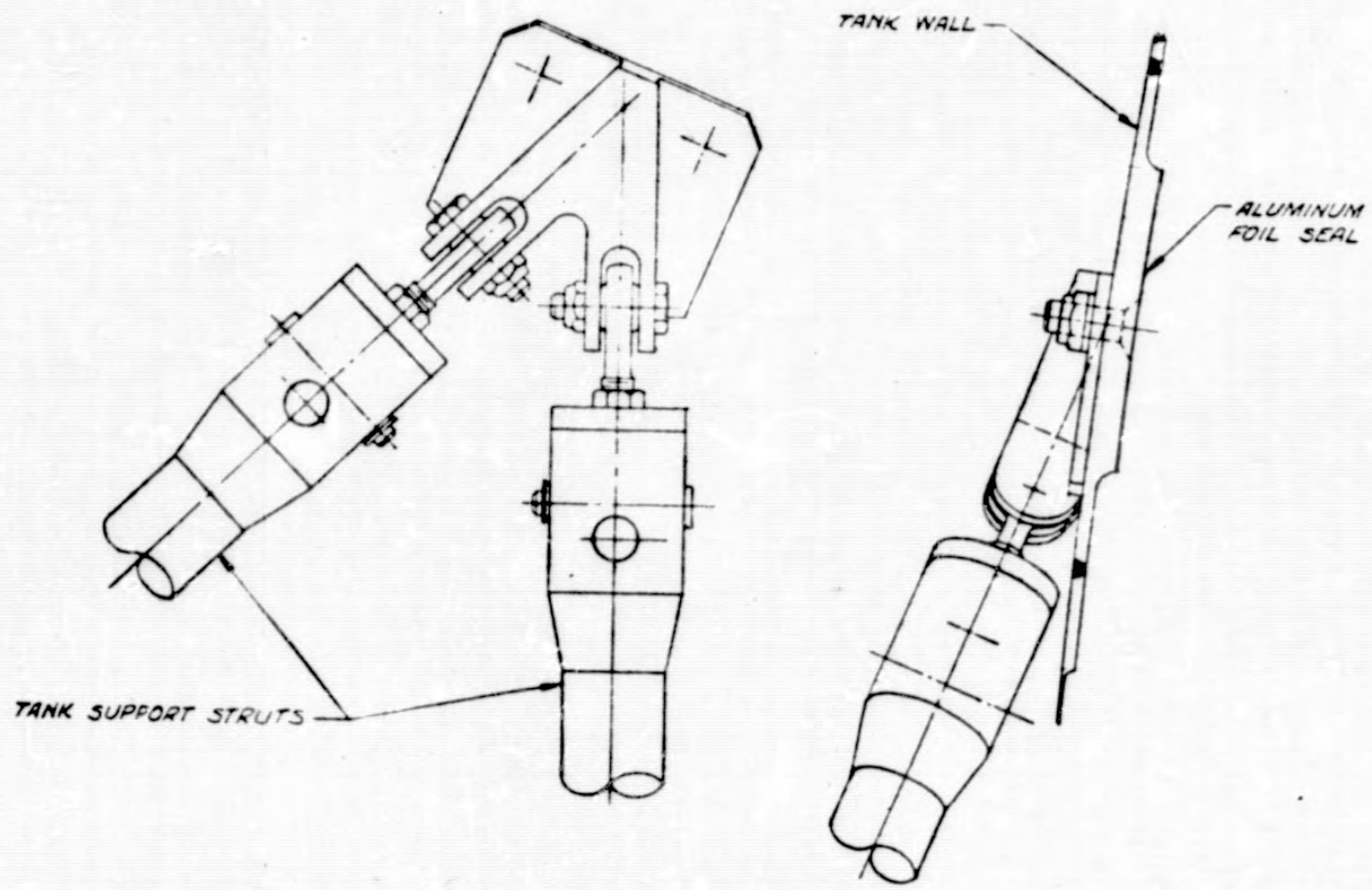
SECTION F-F
SCALE 1/2



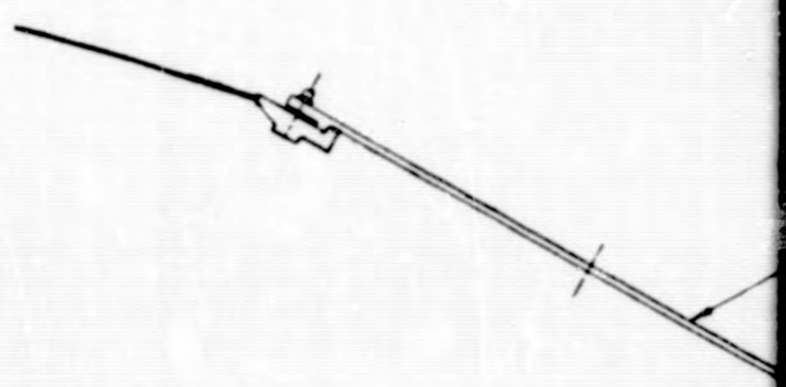
SECTION M-M
SCALE 1/1



SECTION K-K
SCALE 1/1
POINT SENSOR MAST
GUY ATTACH LUG
TYP 3 PLACES

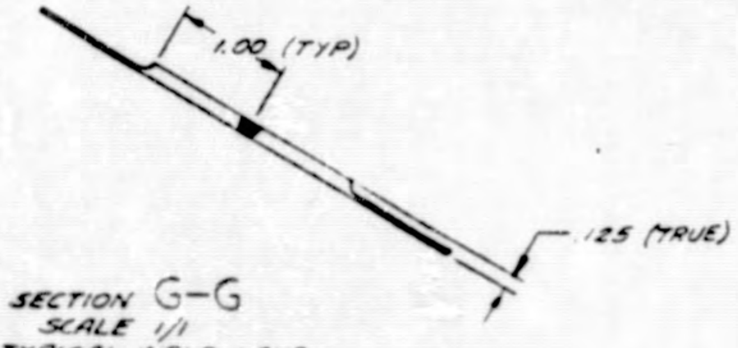


SECTION H-H
SCALE 1/1

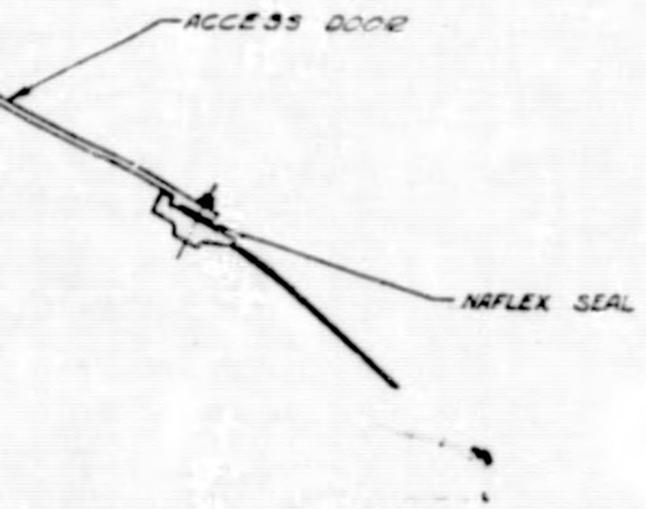


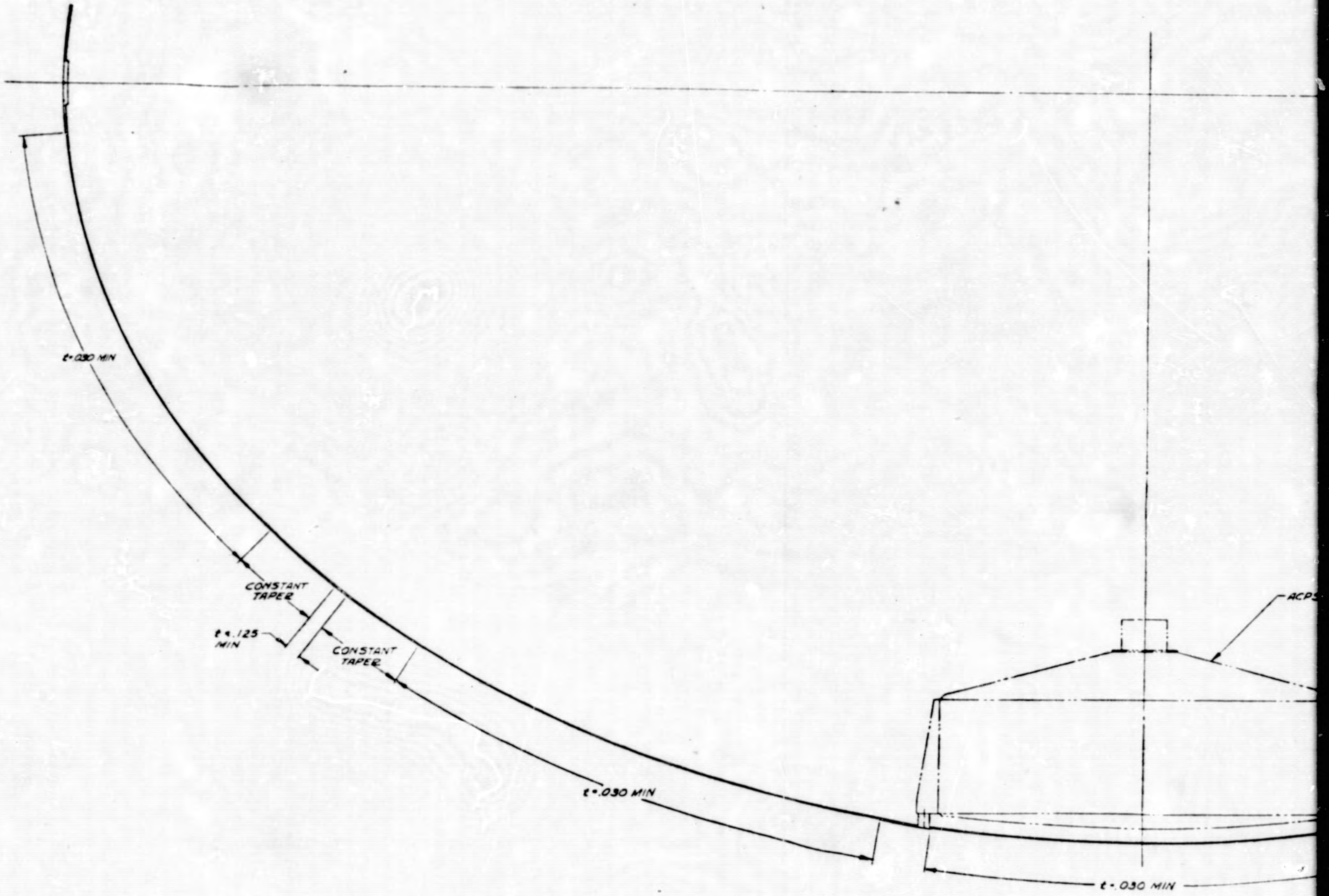
SECTION E-E
ROTATED 135° CCW
SCALE 1/4

F-1



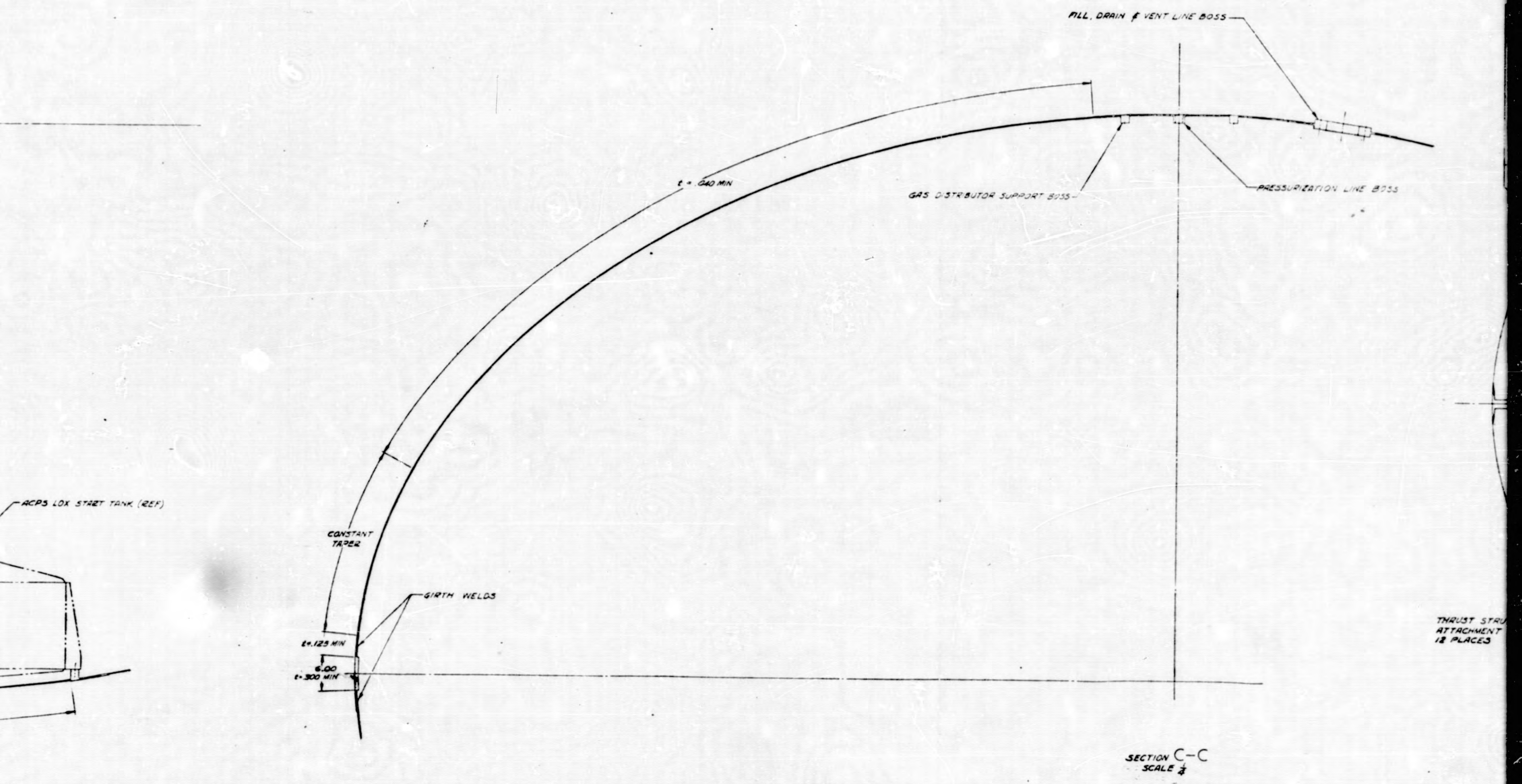
SECTION G-G
SCALE 1/1
TYPICAL WELD LAND
EXCEPT AS SHOWN



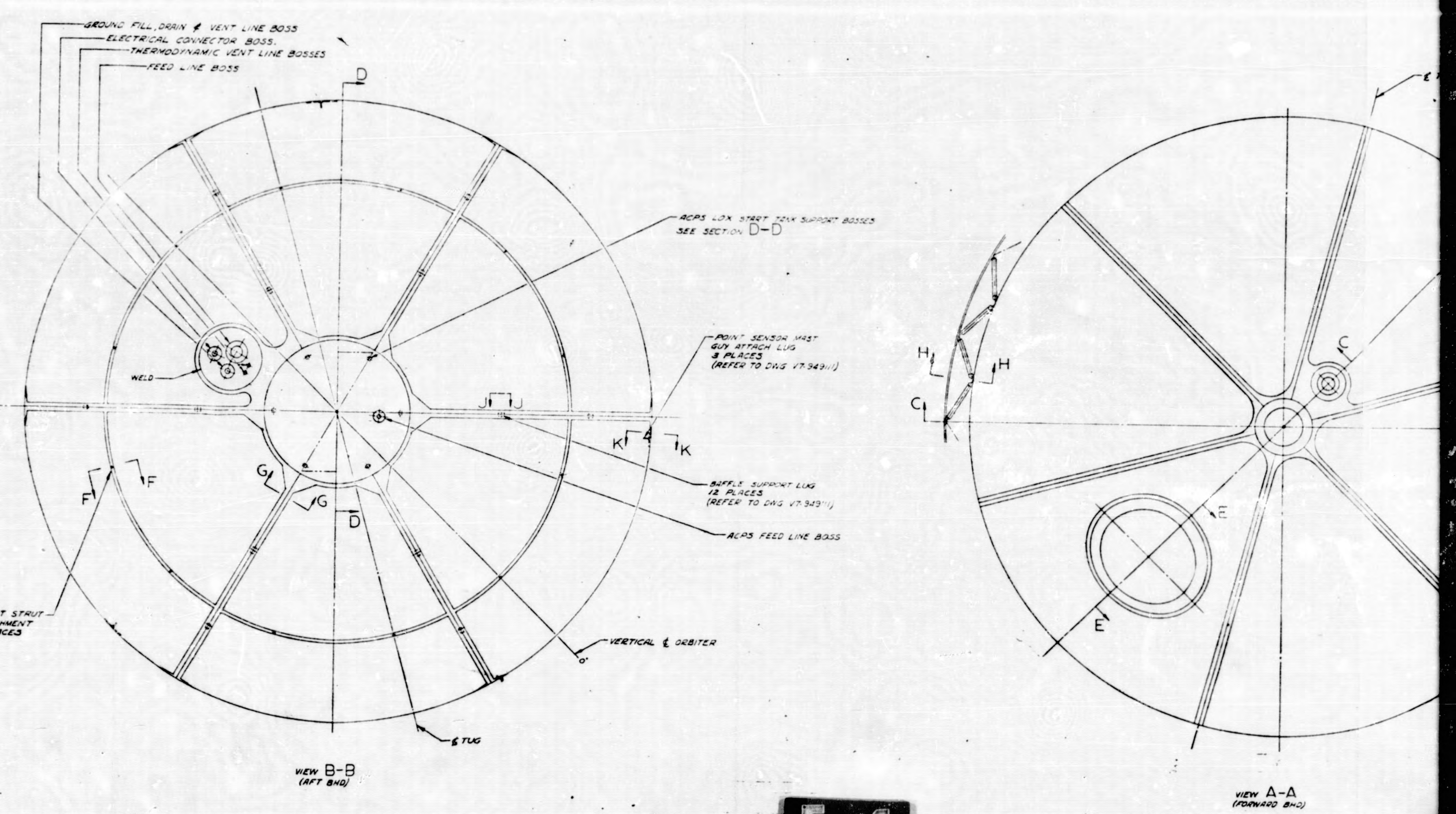


SECTION D-D
(ROTATED 90° CCW)
SCALE 1/4

F-2


FILL, DRAIN & VENT LINE BOSS
t = .040 MIN
GAS DISTRIBUTOR SUPPORT BOSS
PRESSURIZATION LINE BOSS
ACPS LOX START TANK (REF)
CONSTANT TAPER
GIRTH WELDS
t=.125 MIN
6.00
t=.300 MIN
THRUST STRU
ATTACHMENT
12 PLACES
SECTION C-C
SCALE 1/4

F-3



GROUND FILL, DRAIN ¢ VENT LINE BOSS
ELECTRICAL CONNECTOR BOSS.
THERMODYNAMIC VENT LINE BOSSES
FEED LINE BOSS
D
ACPS LOX START TANK SUPPORT BOSSES
SEE SECTION D-D
POINT SENSOR MAST
GUY ATTACH LUG
3 PLACES
(REFER TO DWG V7-949111)
WELD
J J
K A K
F F
G G
D
BAFFLE SUPPORT LUG
12 PLACES
(REFER TO DWG V7-949111)
ACPS FEED LINE BOSS
T STRUT
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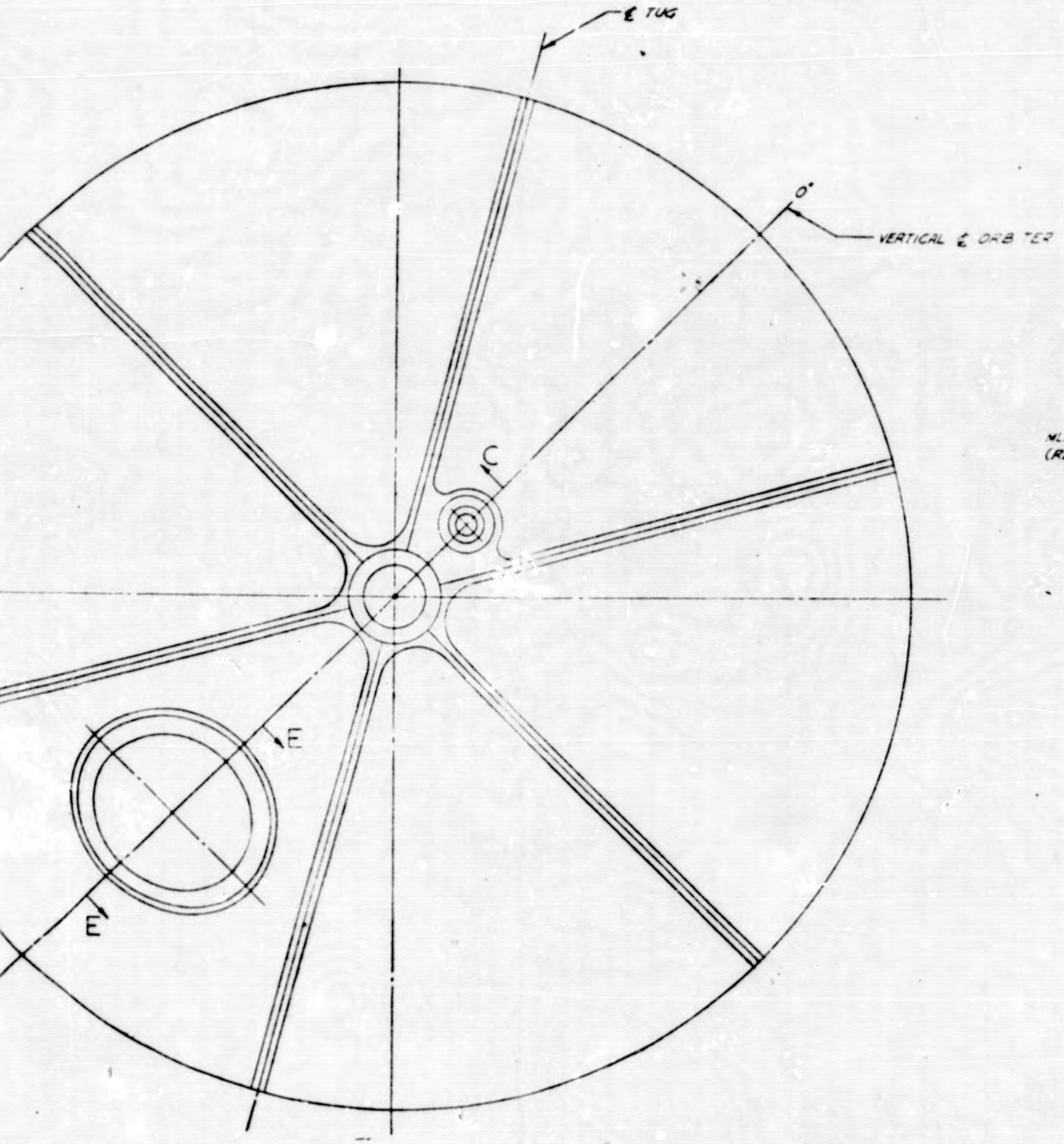
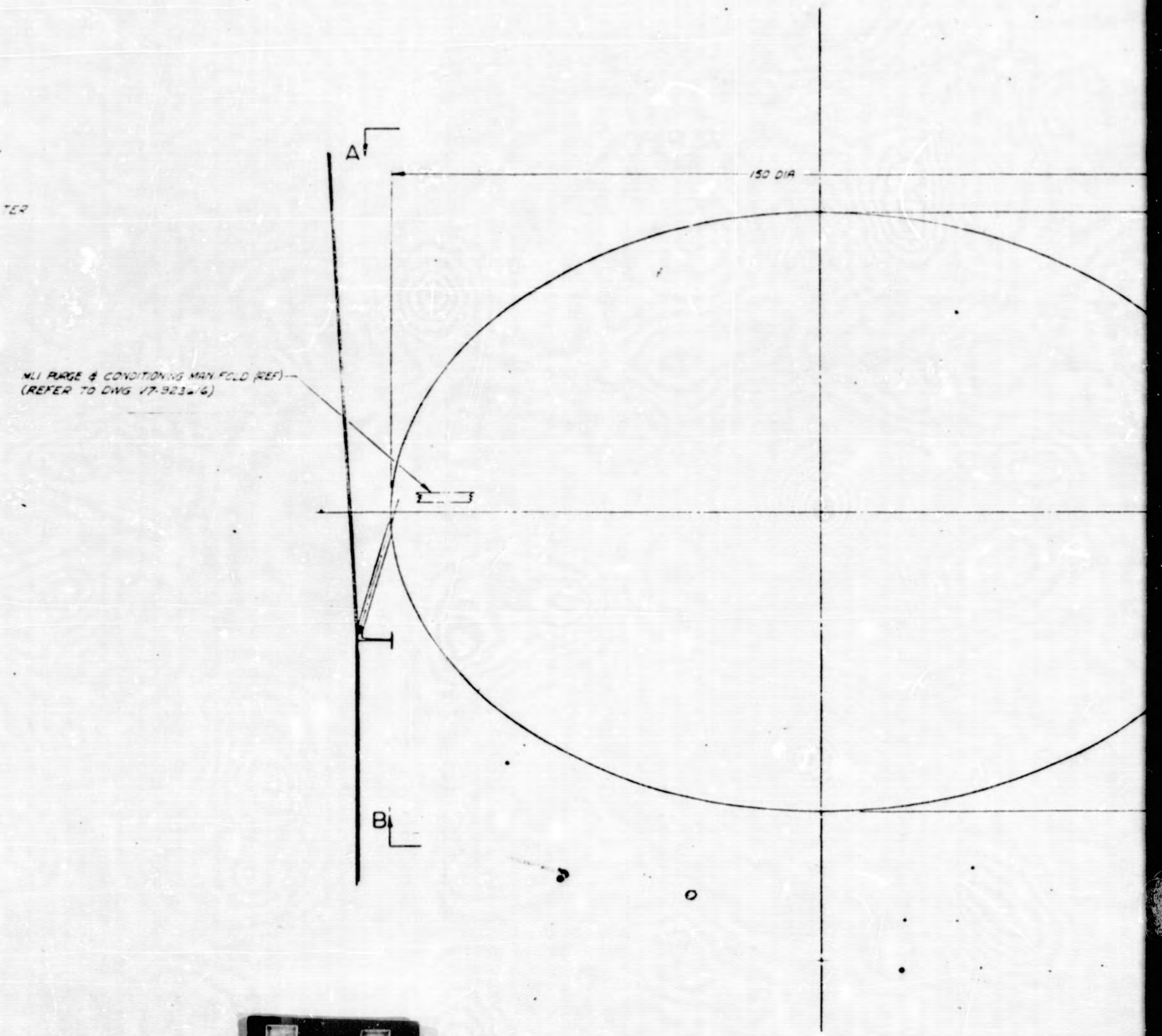
F-4



VIEW A-A
(FORWARD BHD)



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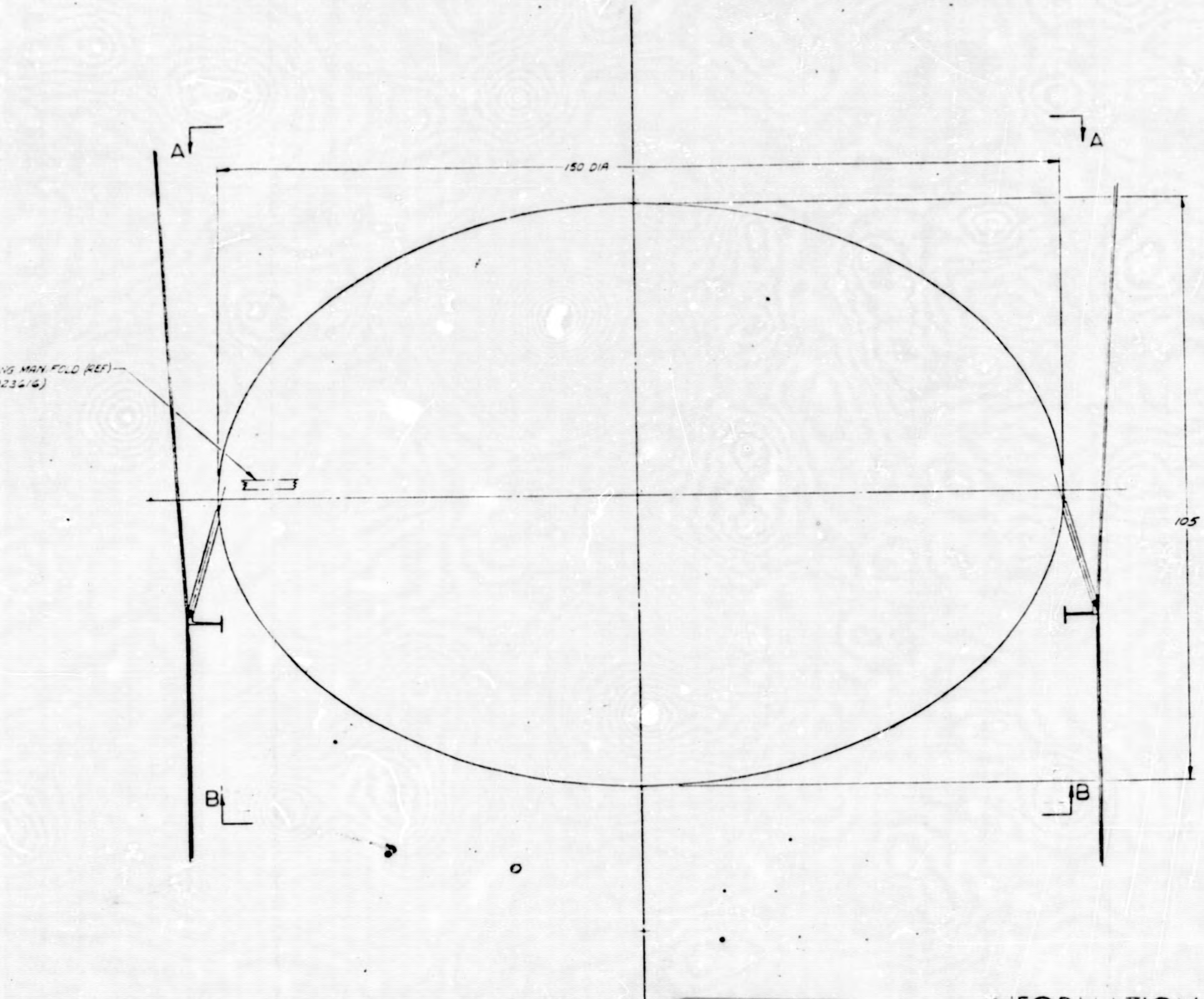

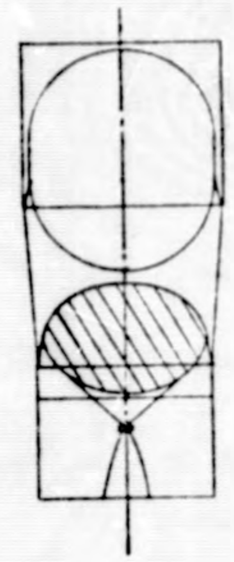

Figure 8.4-2 LOX Tank Structure Tug

2. FOR PROPELLANT ORIENTATION & MANAGEMENT SYSTEMS INSTALLATION SEE DWG V7-349111
1. TANK MATERIAL MB0170-002, 2014-T6 AL ALLOY
NOTES:

F-6

INFORMATION ONLY.

SCALE 1/8 NOTED	DR. ARCHIBALD DATE DEC 13 '71 MODEL	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	LOX TANK STRUCTURE TUG		V7-923609

together and joined at the apex by a circular section which is also butt welded to the gores. The system ports are incorporated into circular sections that are butt welded into the apex circular section and individual gores. The basic wall thickness is .040 inches for the forward bulkhead and .030 inches for the aft bulkhead. A 20-inch diameter access door is located in the forward bulkhead off center and 30° off the Z axis. This door provides access into the LOX tank for installation of equipment. A heavier section at the girth is required for hoop stability and tank wall stresses by the tank support struts. A heavier section, that bands the aft bulkhead, is required to distribute loads induced into the bulkhead by the thrust structure attachment. The APS start tank is mounted inside the LOX tank. It is supported by six integral bosses which are machined into the aft bulkhead dollar weldland.

8.4.2 Structural Analysis

Pressurized Tankage

General

A schematic representation of the Tug pressurized tankage and support system is shown in Figure 8.4-3. Critical loadings occur in the propellant tanks during the Shuttle boost, Orbiter boost, and Tug operation phases of the mission. As shown, the TUG launch configuration is inverted which results in maximum stresses occurring in the LH_2 and LOX tank forward bulkheads during the shuttle boost phases.

Figure 8.4-4 depicts the method by which the baseline Tug tankage configuration was defined. Once the design requirements, pressures, loads, and temperatures were established, the tanks were optimized with respect to a two-stage versus single-stage vent valve system and pressurant gas temperature. Final sizing of the tanks was then accomplished by fracture mechanics analysis considering the vehicle pressure-time-history over a 20 mission life. The basic starting point for the design was the configuration shown in FIGURE 8.4-3. The tankage was not optimized with respect to shape. The tankage configuration shown in Figure 8.4-3 was used throughout the study.

Shuttle payload factors used in the analysis are given in Table 8.2-1. Main propellant tank internal pressure levels used in the analysis are summarized below:

Condition	Pressure (psig)			
	LH_2 Tank		LOX Tank	
	Max	Min	Max	Min
End Boost (Booster Thrust)	24	15	17	15
End Burn (Orbiter Thrust)	24	15	17	15
TUG Operation	24	15	24	15

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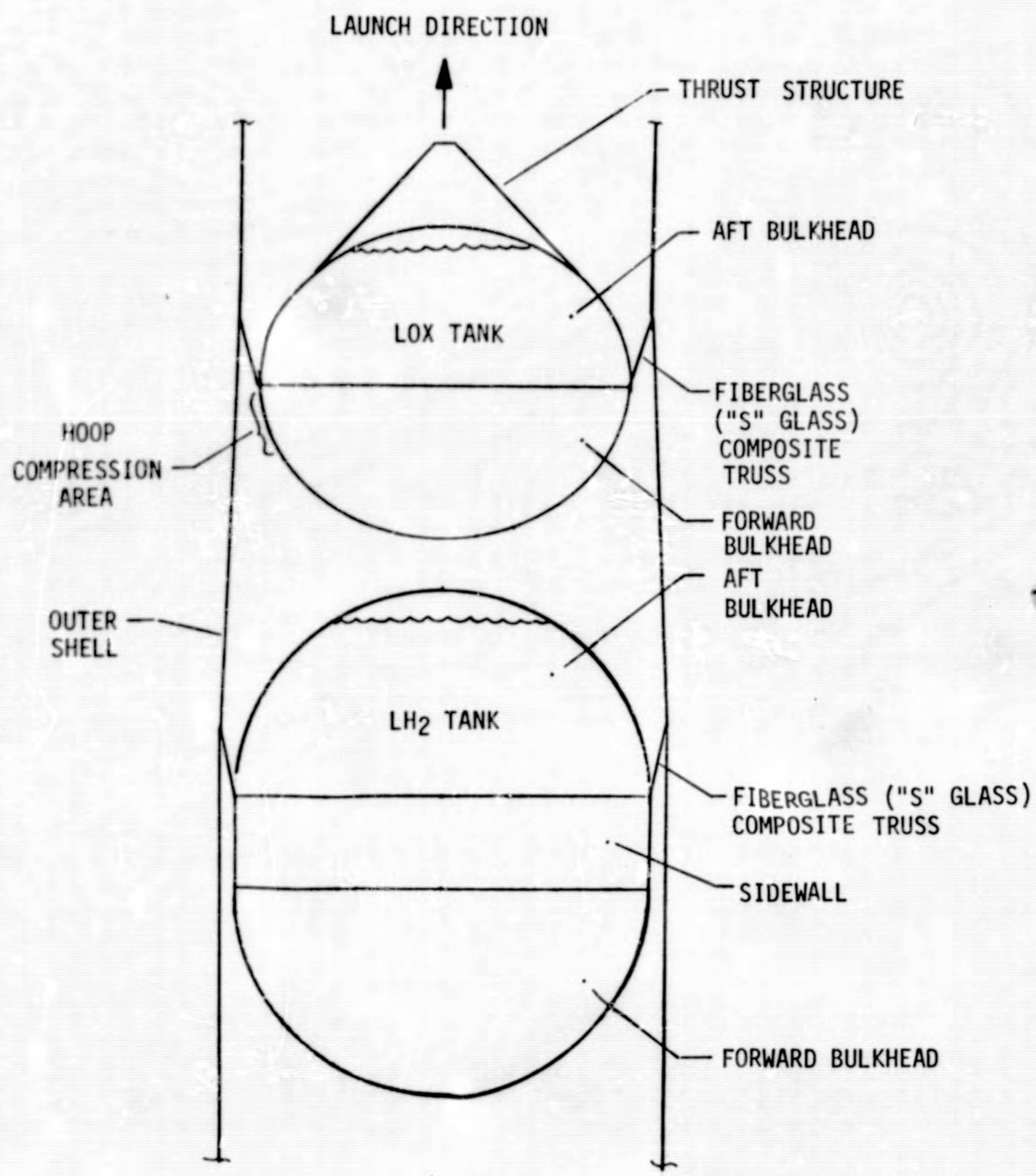

Figure 8.4-3 Pressurized Tankage and Support System ~ Launch Configuration

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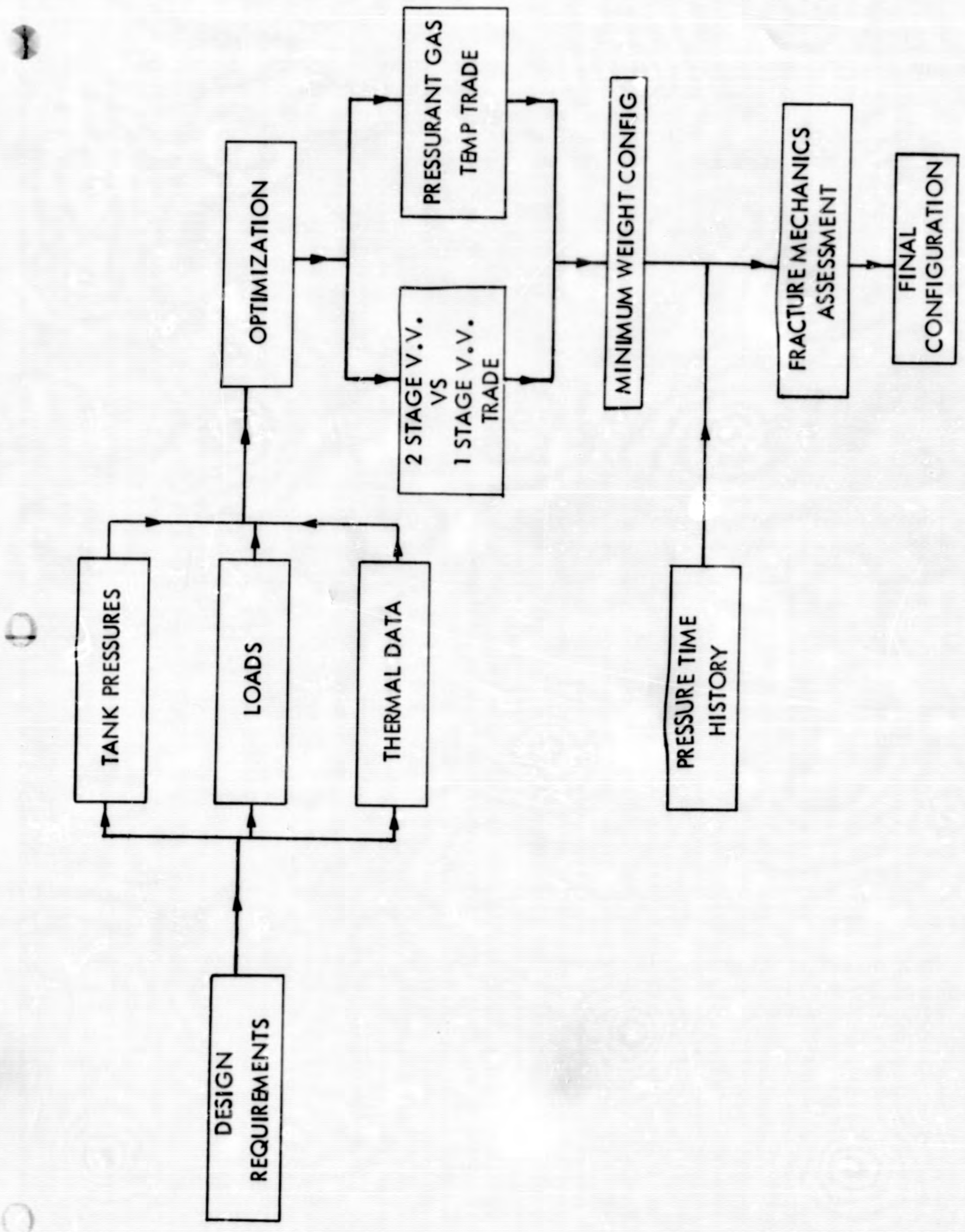

Figure 8.4-4 Configuration Definition ~ Pressurized Tankage

The tanks were designed in accordance with the fracture mechanics guidelines set forth in Reference 7. The fracture mechanics assessment led to the selection of 2014-T6 aluminum as the tankage material. Material selection rationale is described in Paragraph 8.1. The pressure-time histories used in the fracture mechanics assessment are provied in Table 8.1-1 and Figures 8.1-1 and 8.1-2. The TUG was designed for a life of 20 missions.

Factors of safety used are as follows:

Condition	Factor of Safety: Ultimate	Factor of Safety: Yield
Attached to Space Shuttle	1.40	1.10
During TUG Operation	1.25	1.05

Thermal data for the tankage is given in Figure 8.4-5. Material properties and allowables are given in Table 8.1-2.

Method of Analysis

Sizing of Main Propellant Tanks

Two preliminary trade studies were made in order to determine the minimum weight configuration. The first was a study of a two stage vent valve system versus a single stage system for each tank. The main objective of a two stage vent valve system is to reduce the ullage pressure in the tank at a time when the fluid pressures on the forward bulkheads are at a maximum (due to the maximum "g" loading). These occur during the Shuttle boost phases of the mission. The results of the study concluded that a weight saving could be realized by incorporating a two stage vent valve system in the LOX tank. However, no weight advantage could be gained by a two stage system in the LH_2 tank as the most critical condition occurs during the TUG operation phase of the mission when bulkhead temperatures are at a maximum.

The second trade study consisted of assessing the impact on the structural weight of pressurizing the tanks with a "hot" pressurant gas. It was concluded that a temperature above -200 F in the LH_2 tank and above +140 F in the LOX tank would result in a weight penalty to the vehicle.

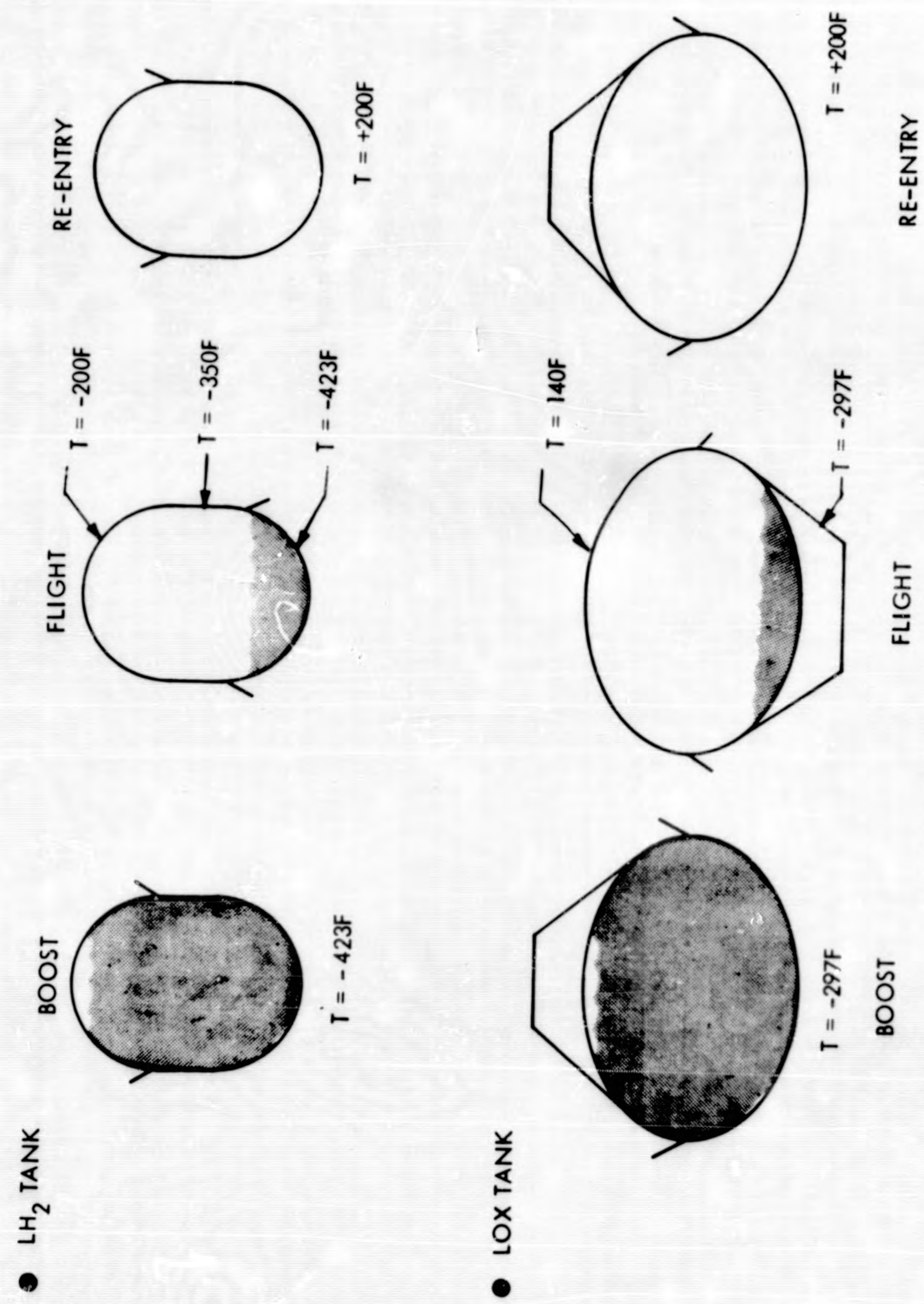

Figure 8.4-5 Main Propellant Tanks ~ Thermal Data

Sizing Procedure

The following procedure was followed in sizing of the main propellant tanks:

1. The tank was initially sized to sustain the basic membrane loads induced as a result of pressure, "g" loading, and temperature. Basic membrane loadings for the LOX tank during its most critical condition are given in Figure 8.4-6 and 8.4-7. IBM Computer Program "PVP," documented in Reference 8.4-7, was used to determine the membrane loadings. The LH_2 tanks is most critical during TUG operation when the temperature reaches a maximum of -200 F in combination with an ullage pressure of 24 psig.

2. The tanks were then assessed from a fracture mechanics standpoint as outlined in Reference 7. The block diagram given in FIGURE 8.4-8, depicts the method used for sizing the tankage to withstand the flaw growth occurring over its 20 mission life cycle. Details of the fracture mechanics analysis are contained in Appendix A.

As shown in Figure 8.4-8 the tankage is initially proof tested at room temperature to a level of 95 percent of yield strength. This is to open up any existing flaws in the welds so they will be more easily detectable. The tankage is then proof tested at cyrogenic temperature to a level of 95 percent of yield strength. Using a plot of gross fracture stress (K_{IC}) versus flaw size, the maximum size flaw (a_i) that could exist in the tank is determined (fracture or tank leakage would occur otherwise). Once this initial flaw size has been determined, the threshold flaw (a_{TH}) is determined using a plot of sustained stress threshold versus flaw size. The threshold flaw (a_{TH}) being that size flaw which will grow under a sustained operating stress level. The difference between the threshold flaw and the initial flaw (Δ a) is the allowable flaw growth in the pressure vessel.

$$\Delta a = a_{TH} - a_i \qquad (1)$$

In the example shown in Figure 8.4-8, the initial flaw was determined to be about 65 percent of the tank thickness. However, it should be noted that the initial flaw size is not a constant percentage of the thickness but is a function of thickness, stress level, and temperature. Therefore, the precise solution of the tank thickness is a iterative process.

Once the allowable flaw growth has been determined, the flaw growth during the 20 mission life may be determined for each region of the tank.

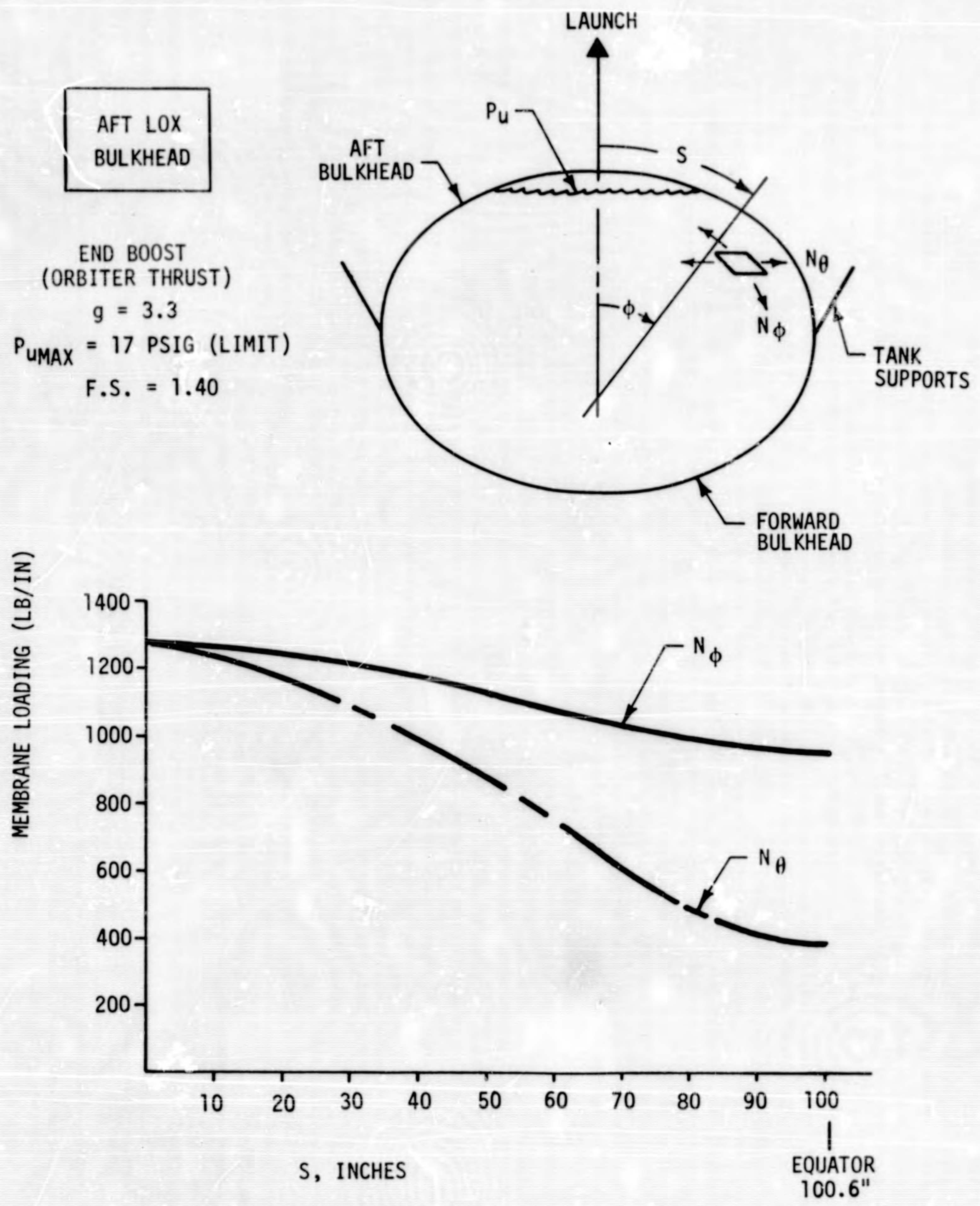

Figure 8.4-6 Membrane Loading LOX Aft Bulkhead

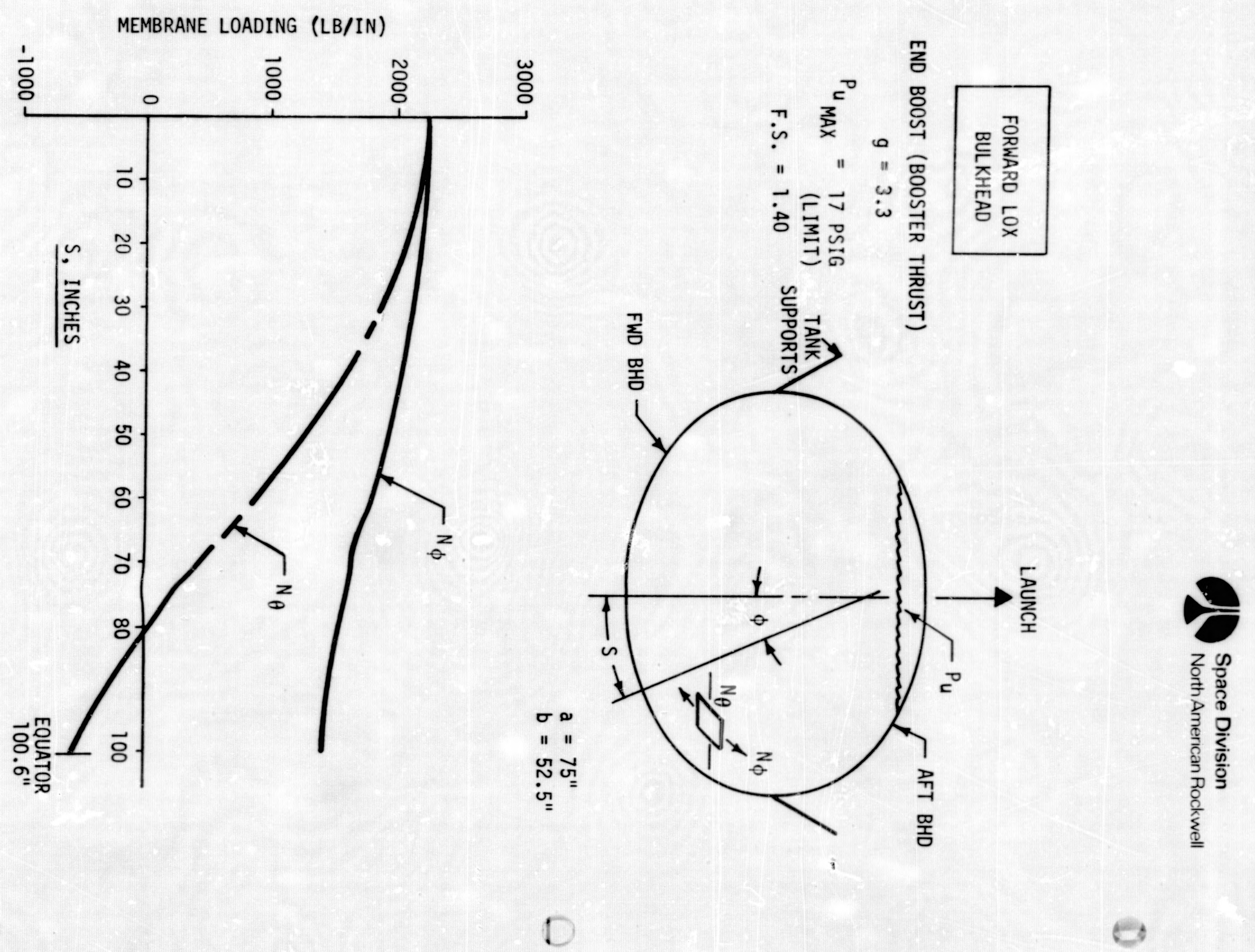

Figure 8.4-7 Membrane Loading Forward LOX Bulkhead

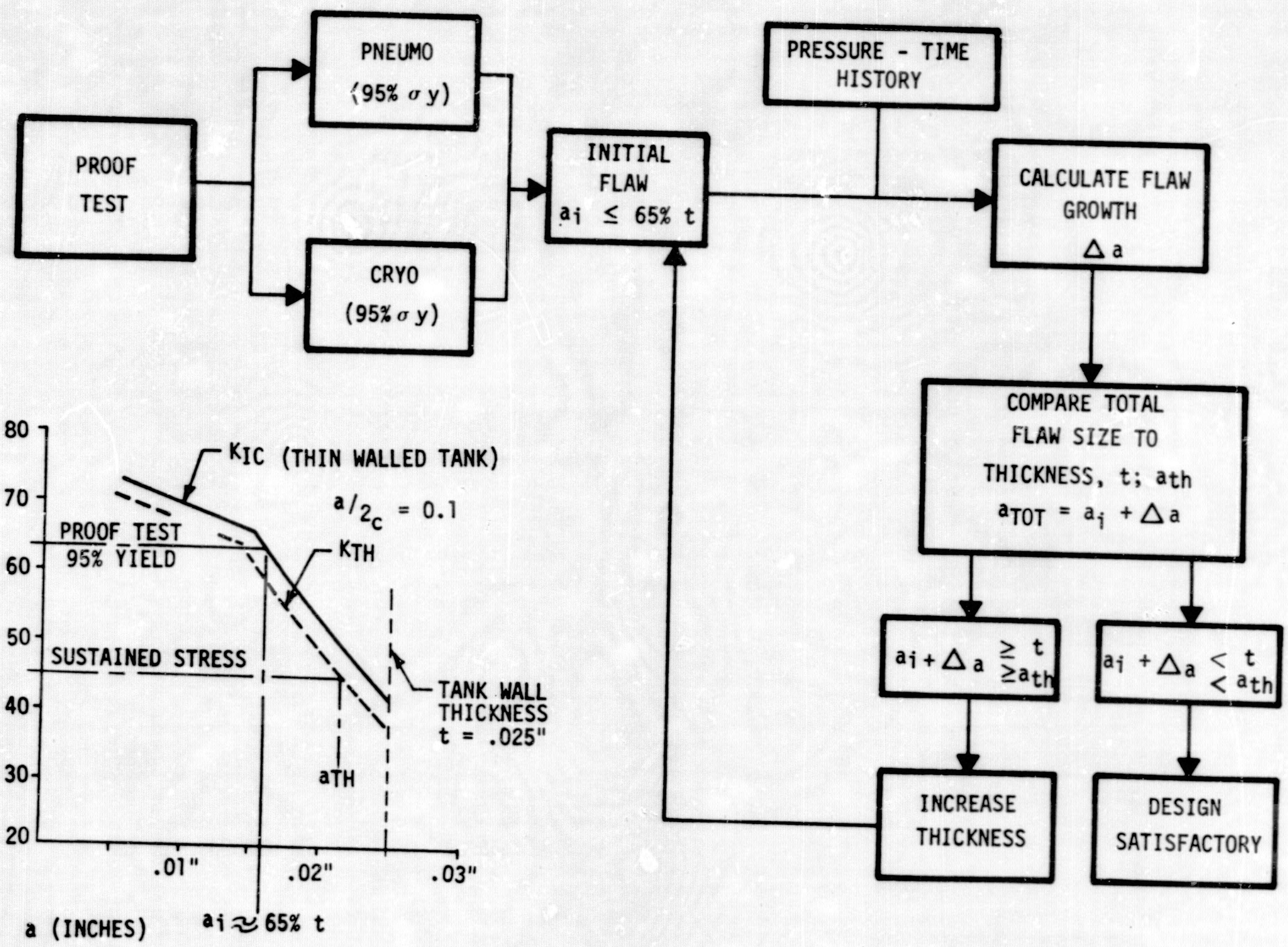

Figure 8.4-8 Fracture Mechanics Assessment ~ Main Propellant Tanks

A computer program was developed on the OS/360 Remote Access Computing System (RAX) in order to determine the flaw growth in various areas of the pressurized tankage over the entire 20 mission life. The results of these analyses are summarized below:

		t_{Min} (In)	a_i (In)	$\frac{da}{dN}$ (In)	a_{TOT} (In)	a_{TH} (In)
LH_2	Bulkheads	0.020	0.012	0.0018	0.014	0.015
TANK	Sidewall	0.045	0.018	0.0012	0.020	0.027
LOX	Forward Bulkhead	0.040	0.025	0.0010	0.026	0.029
TANK	Aft Bulkhead	0.030	0.016	0.0016	0.018	0.023

The proof test levels required to screen-out the initial flaw sizes are as follows:

	PRESSURE (PSIG)	
	LH_2 TANK	LOX TANK
PNEUMO	26.2	30.8
CRYOPROOF	33.6	35.0

These pressure levels are based on a maximum stress level of 95 percent of tensile yield strength.

Hoop Stability ~ LOX Forward Bulkhead Equatorial Region

During the shuttle and orbiter boost phases of the Tug mission, approximately 20 inches of the forward bulkhead is subjected to hoop compression loading as shown in Figure 8.4-7. These compression loads are developed as a result of the "g" loading of the fluid on the shell in conjunction with the ellipsoidal shape of the bulkhead.

The critical stability case occurs when the LOX tank ullage pressure is a minimum (15 psig) and the acceleration is a maximum (3.3 g's); the end boost (booster thrust) condition. The factor of safety is applied only to the LOX head pressure component, i.e.,

$$P_{TOT} = 15.0 + 1.4\ (3.3\ (\text{LOX head}))$$

This creates the most severe case of largest hoop compression loads in combination with the smallest meridional (stabilizing) loads.

In performing the analysis for stability of this area of the forward bulkhead, the curvature in the hoop direction is neglected and the analysis is performed on a monocoque cylinder of equivalent thickness, with radius equal to the meridional radius of curvature of the ellipsoid, and subjected to axial compression and internal pressure loadings. The method of analysis is contained in Section 9.22.01, Reference 5.

Results of the analysis determined a constant taper in thickness (0.125" to 0.030") from the equator along an arc length of 20 inches would sustain the compressive loading in the shell.

Summary of Maximum Operating Stresses

The maximum operating stress levels for the propellant tanks are summarized as follows:

	Region	t_{min} (in.)	Max. Operating Stress (psi)	Condition
LH_2 Tank	Bulkhead	0.020	50400	During TUG Operation
	Sidewall	0.045	44800	Booster Boost
LOX Tank	Forward Bulkhead	0.040	41000	Booster Boost
	Aft Bulkhead	0.030	43000	During TUG Operation

Tankage Support System

General

The propellant tankage composite truss support system is shown schematically in Figure 8.4-3. Details of the support system are given in Figure 8.3-5. The critical condition for the support system occurs at end boost (booster thrust) when "g" levels are at a maximum. Shuttle payload load factors used in the analysis are given in Table 8.2-1. Material properties are given in Table 8.1-3.

"S" Glass Composite Struts

Critical loading for the tankage support struts occurs when the struts are loaded in tension. Maximum strut loadings and capabilities are summarized as follows.

	t_{WALL} (In)	DIA (In)	F_{tu} (psi)	TENSION CAPABILITY (lbs)	APPLIED LOADING (ULT) (lbs)
LH_2 TANK	0.012	1.00	220,000	6200	2100
LOX TANK	0.024	0.75	220,000	16700	10200

It should be noted that the above capabilities are for the basic strut and that the critical area will most likely be the strut to end fitting attachment joint. The capability of this joint can be increased to desired levels by increasing the strut end bearing area, and by increasing the number or size of fasteners without a significant increase in structural weight.

Tank to Strut Joint

The propellant tankage to strut joint is also critical during the end boost (booster thrust) condition when maximum otward radial loading occurs on the tank shell. The tank shell region at the joint is analyzed as a ring subjected to equal radial forces, equally spaced (Reference 10, case 9, Table VIII).

In both tanks, the longitudinal welds are critical. These welds are located midway between the concentrated loads at three locations in each tank. Results of the analyses are as follows:

	WELD STRESSES (ULT)			ALLOWABLES		
	Hoop (psi)	Bending (psi)	TEMP (°F)	F_{tu} (psi)	F_b (psi)	FACTOR OF SAFETY
LH_2 TANK	28700	8200	-423	41000	-----	1.55
LOX TANK	9800	42800	-297	37000	55500	1.40

8.5 DOCKING SUBSYSTEM

Figure 8.5-1 is a schematic of the Space Tug docking subsystem. Docking assemblies are provided at two locations; the forward skirt for docking with the payload, and at the aft skirt/Space Shuttle adapter interface for docking in preparation for loading the Tug in the Shuttle orbiter and reentry. Presented here are descriptions of the two docking assemblies, loads analysis, and structural analysis of the probe support structure.

Tug/Adapter Docking

Docking of the Tug to the adapter may be accomplished when the two mating surfaces are within approximately 1.5 ft and the combined axial and rotational misalignment of mating point near the periphery is within 1.0 ft. Three

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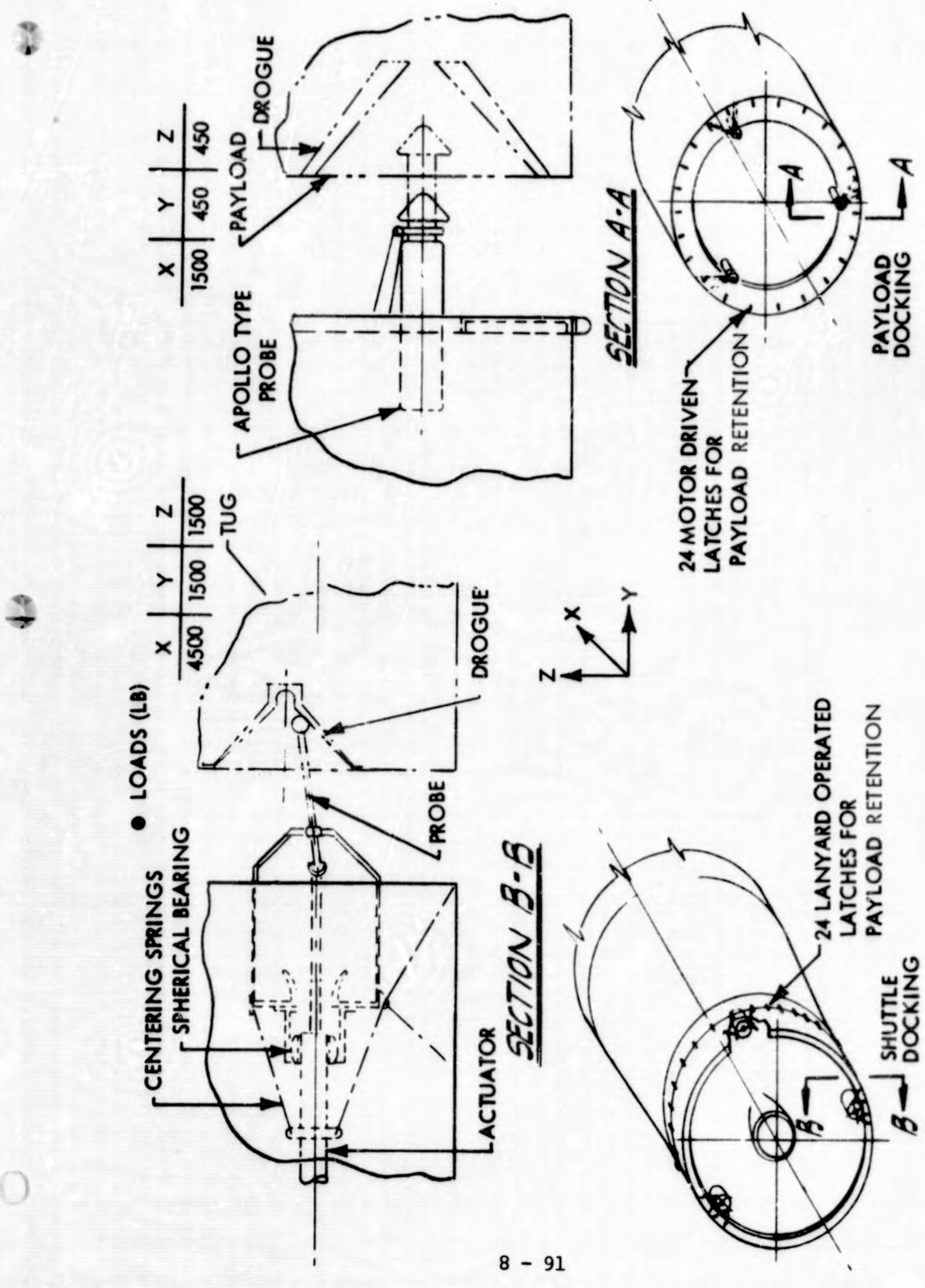


Figure 8.5-1 Docking Subsystem

actuated probes are extended axially from the adapter to engage three recessed cone shape drogues mounted on the aft face of the Tug structure. When extended, the probe arm is free to deflect laterally under slight pressure and to be guided into the drogue latch socket. As the three probes are retracted, they are guided into parallel axial alignment orienting the Tug about all three axes. The final inch of movement is parallel to the body centerline snubbing the interface surfaces together.

To accomplish this, the probe arm and actuator strut are connected by a ball joint. The actuator and probe are each supported by a spherical bearing. The probe slides inside the spherical support. A spherical surface collar mounted near the probe/actuator joint engages a bell shape fairlead fitting when the actuator is retracted, forcing axial alignment of the probe. A sperical ball with expanding segments on the probe end and a matching drogue receptacle allows for angular misalignment on initial contact. Latching is activated by the probe.

Centering bungees with damping provisions are attached to the actuator and keep the extended probe in axial alignment until a side force is encountered.

A spring and dampening device allows the probe to telescope and provide axial attenuation.

A gas operated actuator is assumed; however, a hydraulic system, if available on the orbiter, would provide better control and more flexibility.

The main advantage of the jointed probe is that initial engagement with the drogue may be made without translation of the vehicle mass. Alignment and mating is then controlled by the probe mechanism.

The heel lines of the channel section ring frames at the aft end of the aft skirt and the forward end of the adapter butted together from the Tug-Adapter interface. Twenty-four latches are located externally around the perimeter of the interface. Sixteen of the twenty-four latches are folded away after initial deployment. The remaining eight independently actuated latches are used for docking to secure the Tug for reentry.

In addition to the frame caps, fittings at the eight lock/release points are butted together at the parting plane and are pre-loaded in compression by the overcenter actuated latch. All other fixed structure is recessed from the parting plane to prevent contact. The 16 release latches have eccentric compression links which grip both sides of the Tug fitting flanges providing for both tension and compression during launch to EOS orbit. Shear is transmitted through the serrated connecting link to serrated pads on each side of the parting plane. The release latches are installed in the assembled locked position with the two bodies mated. The serrated pads are aligned and shimmed to mesh with the connecting links. In addition to the overcenter pre-loaded forces on the link, a safety latch on the release arm prevents disengagement of the serrations. For re-mating the latches are manually engaged. Operationally, several latches are released by a single actuator through lanyards attached to the release arms. The initial movement releases the safety latch.

Additional movement torques the connecting link and "dogs" off the fitting flange and stows to clear the separation plane for docking. The eight lock/release latches are individually actuated through an overcenter linkage system, which provides for a longitudinal clamping force in the final increment of movement.

Tug/Payload Docking

For Tug-to-payload docking, a system of three probes, similar in concept to those used for Apollo, are shown. These probes would be greatly simplified from the actual Apollo probes because many of the complex requirements for Apollo-LEM docking would not exist for the permanently fixed probes that would be used for Tug-to-payload docking. Three off-center probes are required because the forward end of the main LH_2 tank is too close to the forward interface plane to allow the installation of a probe assembly on the Tug centerline. For docking acquisition the probe heads are extended ten inches. Activation is accomplished with an internal pneumatic system. Helium containers are carried in the probe housing. Electrically initiated pyrotechnic devices release the helium to serve for actuation and attenuation in docking.

Hard latching is accomplished by latching fingers spaced around the periphery of the forward interface. A latch is located at each 15 degree interval to assure uniform distribution of loading. The latching fingers are translated by gearing and are electrically actuated. Redundancy might be obtained by use of dual, parallel drive motors or by use of override pyrotechnic devices. The latch fingers are beveled and fit against matching beveled surfaces on the payload.

8.5.1 Docking Loads Analysis

Mission-Dictated Docking Modes

There are two basic mission-dictated docking modes in TUG operations. The first is the retrieval mission in which the Tug docks to a payload of up to 4160 pounds for powered return flight to rendezvous with the EOS. The second is the return mission in which the Tug docks, with or without payload, with EOS. According to various mission purposes, the Tug interface dockings are further classified and presented in Table 8.5-1.

Dynamic Analysis

A computer program was prepared for the determination of orbital docking loads. Four docking conditions were considered: demate from the EOS, docking with the payload, demate from the payload, and docking with the EOS. The docking analyses included are:

(a) Initial contact from the design initial conditions to determine the linear velocity, angular velocity, impulsive force and impulsive torque after collision.

Table 8.5-1 Tug Interface Docking

EOS = EARTH ORBIT SHUTTLE EGO = EQUATORIAL GEOSYNCHRONOUS ORBIT, T = TUG, P = PAYLOAD

MISSION	DEMATE FROM EOS	PAYLOAD DELIVERY	DOCKING: PAYLOAD	DOCKING: EOS
FROM EOS TO EGO & RETURN TO EOS DELIVER OR RETRIEVE 3000 LB PAYLOAD	FWD T P AFT	T P		FWD T AFT
	T P	T P	T P	T P
FROM EOS TO EGO DELIVER 8060 LB PAYLOAD & RETURN	T P	T P		T
FROM EOS TO EGO RETRIEVE 4160 LB PAYLOAD & RETURN	T		T P	T P

(b) Continuous maneuver from the initial phase to determine the angular velocity of the EOS or Tug prior to latching, and to determine the total attitude change of the EOS or Tug during the maneuvering phase.

(c) Final latching to determine the angular velocity of the combined system after latching, and latching forces and torques.

Since the process of demating involves only simple design of releasing or ejecting mechanisms, the dynamic loads can be readily determined from the designed mechanism. The demating reaction loads can be designed to a negligible degree of magnitude in comparison with the docking loads. Therefore, no analysis is necessary at this time. The three-phases docking analyses with or without control forces have been programmed for the digital computer.

For the prescribed configuration of the Tug, Space Shuttle Orbiter, Mark 2 (176A), and various cylinder-shape payloads, the preliminary results are tabulated in Table 8.5-2.

Docking Loads

The dynamic equations for the docking mechanism itself could be included in the continuous maneuver phase of the docking dynamic analysis, but it is simpler to consider the dynamic equations of the docking mechanism separately. The active and inactive vehicles can be considered as two masses. The two masses are connected with springs and dampers to represent the maneuver conditions. From such a setup, the displacement and force equations are obtained. The approximate force equation has the following form:

$$\ddot{X}_T = e^{\alpha t} \left\{ \ddot{X}_{TO} \left(\text{Cox } wt + \frac{\alpha^2 - w^2}{2\alpha w} \sin wt \right) \right\}$$

where

$$\alpha + iw = \frac{1}{2M_T M_S} \left\{ - D(M_T + M_S) \pm \left[-D^2 (M_T + M_S)^2 + K M_T M_S (M_T + M_S) \right]^{1/2} \right\}$$

D = Damping constant in lb-sec/ft

K = Spring constant in lb/ft

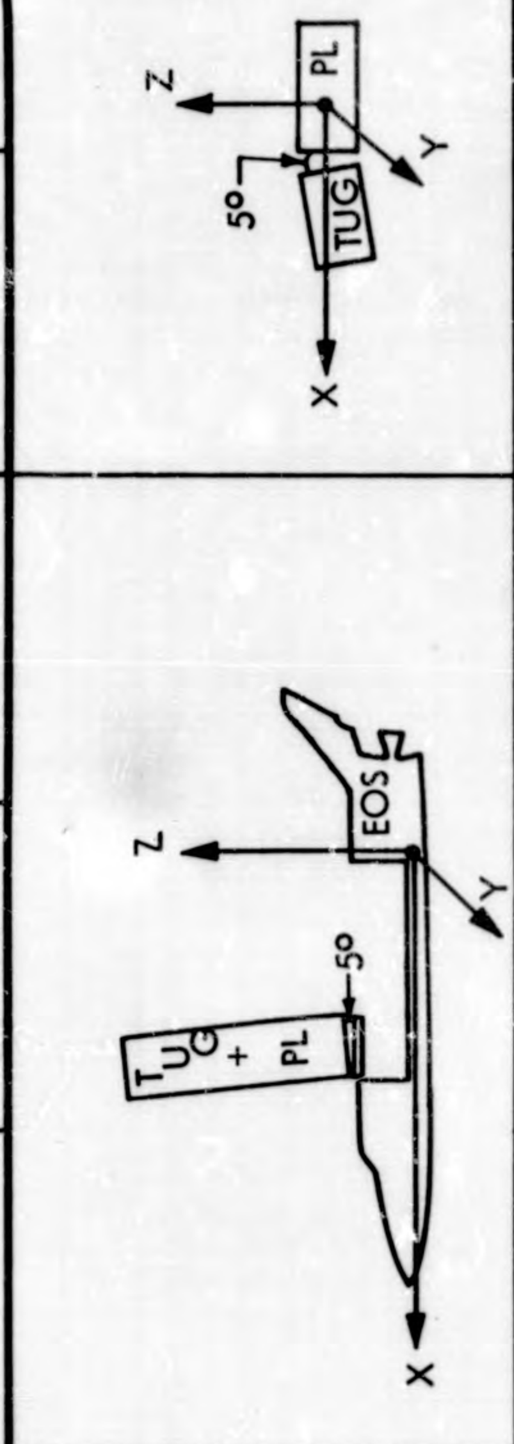
$$\ddot{X}_{to} \cong \frac{D(\dot{X}_{TO} - \dot{X}_{XO})}{M_r}$$

M_T = Mass of Tug

M_S = Mass of EOS

$\dot{X}_{TO} - \dot{X}_{SO}$ = Initial relative velocity

Table 8.5-2 Docking Dynamics

CONFIGURATION		TUG + PL (4160 LBS) / EOS	TUG + PL (3000 LBS) / EOS	TUG/EOS	TUG/PL (4160 LBS)	TUG/PL (3000 LBS)
INITIAL CONDITIONS: ORBITAL SPEED = 25690 FT/SEC; REL CONTACT VEL = 1.0 FT/SEC (LONG) 0.3 FT/SEC (LAT); REL ANGULAR VEL = 0.5°/SEC; DOCKING ARM = 2.0 FT						
CONTACT	IMPULSIVE FORCE (LB-SEC) X	20	11	-13	104	78
	IMPULSIVE FORCE (LB-SEC) Z	292	263	180	5	4
	IMPULSIVE TORQUE (FT-LB-SEC) Y	2920	2350	1080	710	530
MANEUVER	ANGULAR VELOCITY (DEG/SEC)	0.14	0.13	0.09	0.27	0.26
	ATTITUDE ANGLE (DEG)	13	12	9	26	25
LATCH	FINAL ANGULAR VEL (DEG/SEC)	0.10	0.10	0.08	0.07	0.08
	IMPULSIVE FORCE (LB-SEC)	58	52	35	22	16

The results of Tug/EOS and Tug/payload docking force time history are shown in Figures 8.5-2, 8.5-3 and 8.5-4 for the Tug plus 4160 lb payload docking with the EOS, Tug alone docking with EOS, and the Tug docking with the 4160 lb payload, respectively. Each figure is a parametric presentation which shows the variation of the docking force for low, medium, and high damping and soft, medium and hard springs. The figures may be used for approximate design loads for various docking concepts.

The following preliminary design docking loads were selected:

Docking Mode	Axial Load (lb)	Lateral Load (lb)	Normal Load (lb)
Tug/EOS	4500	1500	1500
Tug/Payload	1500	500	500

These loads are conservative enough to allow a wide variation of docking concepts.

8.5.2 Docking Probe Structural Assessment

There are three equally-spaced docking probe mounts on both the forward skirt and the adapter. Graphite epoxy tubes are the main supporting members, and these are designed using the classical equations for orthotropic cylinders developed by Dow and Rosen (Reference 8.7-3), modified by empirical factors from test data. The tubes are fabricated in six layers (four longitudinal and two circumferential). The critical loading for both forward and aft docking systems occurs during docking maneuvers.

Forward Docking Probe

The forward docking probe mounts are located at three equally-spaced positions on the forward skirt. Each mount consists of two rear tubes attached to the Station 420.5 frame, plus a yoke connected to the Station 452 frame. Using a 1.25 factor of safety, each mount must withstand simultaneous ultimate loads of 1875 pounds axially (longitudinally) are 625 pounds laterally (Reference Page 8-118). Design reactions at the supports are shown in Figure 8.5-5. The graphite epoxy tubes are two inches in diameter and 0.036 inches thick. The aluminum I-section yoke tapers in height and width, with cap thicknesses of 0.10 inches and a web thickness of 0.06 inches.

Aft Docking Probe

The aft docking probe mounts are located at three equally-spaced positions on the adapter. Each mount consists of two rear tubes attached to the Station 99 frame, one front tube attached to the Station 126 frame, and a shear web which cradles the probe housing and is also tied to the Station 126 frame. Using a 1.40 factor of safety, each mount must withstand simultaneous ultimate

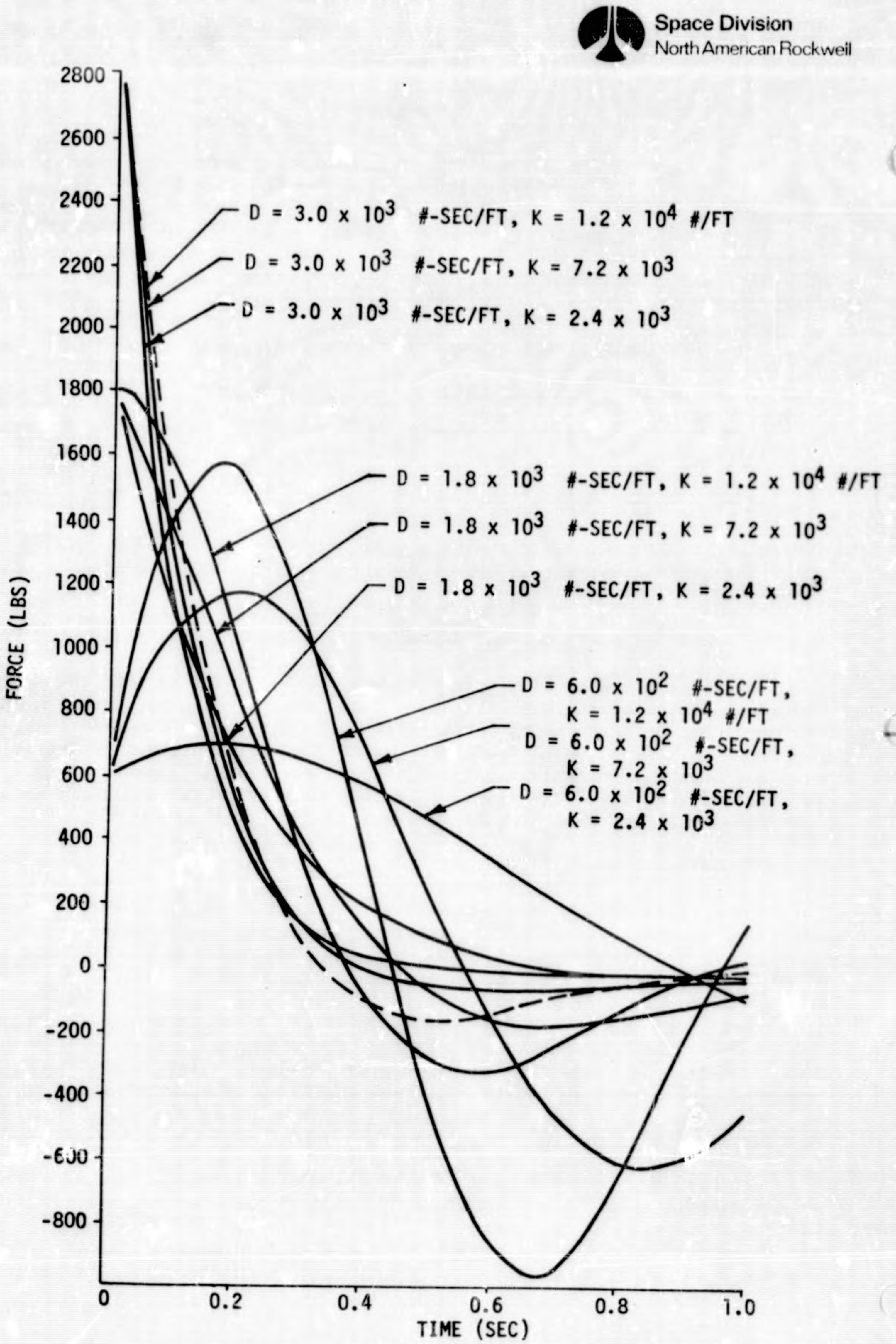


Figure 8.5-2 Tug + 4160 Lb Docking With Shuttle Orbiter
$M_{TUG} = 3.106 \times 10^2$ Slug, $M_{EOS} = 4.995 \times 10^3$ Slug, $V_o = 1.0$ Ft/Sec

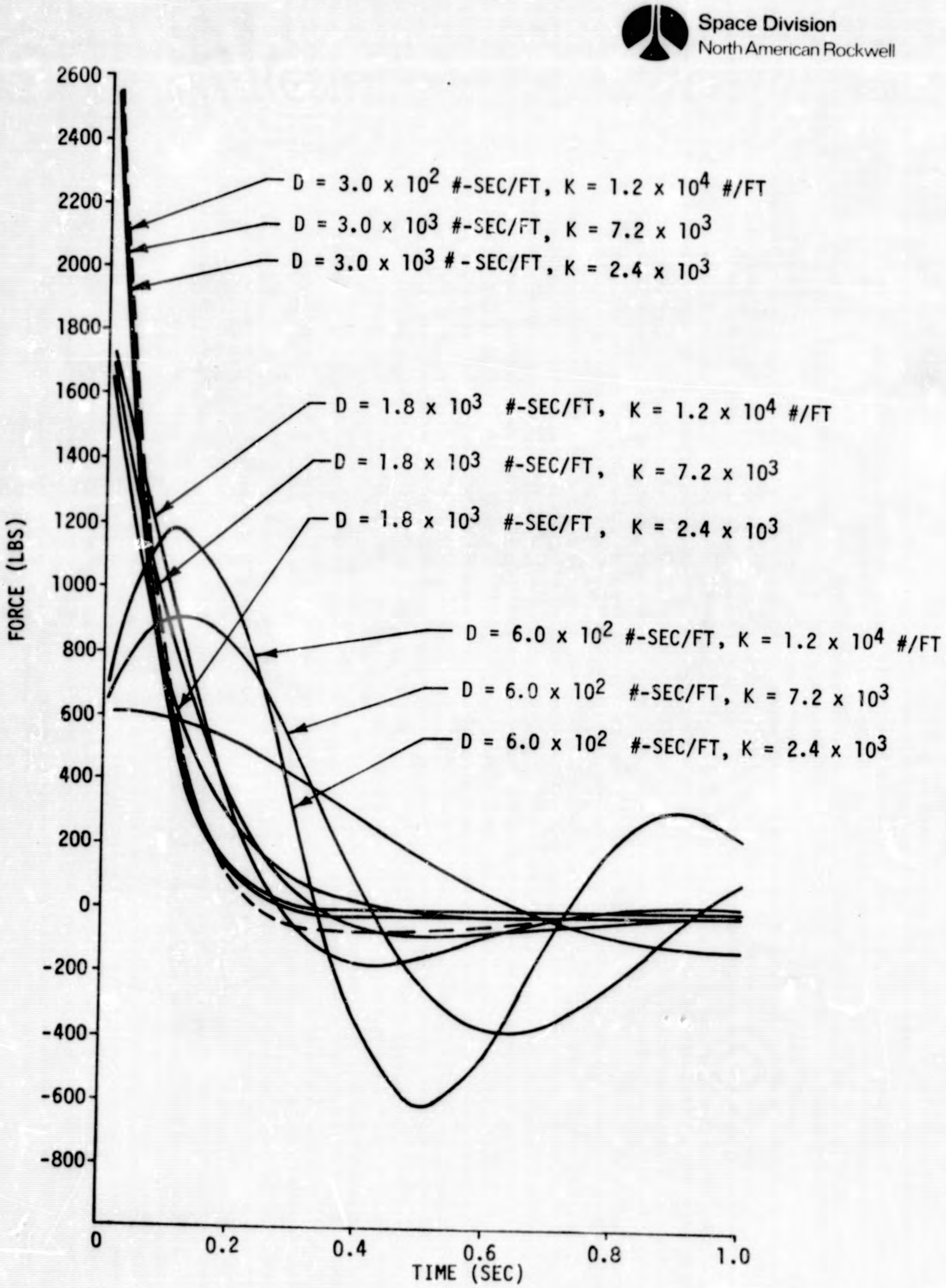


Figure 8.5-3 Tug W/O Payload Docking With Shuttle Orbiter
$M_{TUG} = 1.829 \times 10^2$ Slug, $M_{EOS} = 4.995 \times 10^3$ Slug, $V_o = 1.0$ Ft/Sec

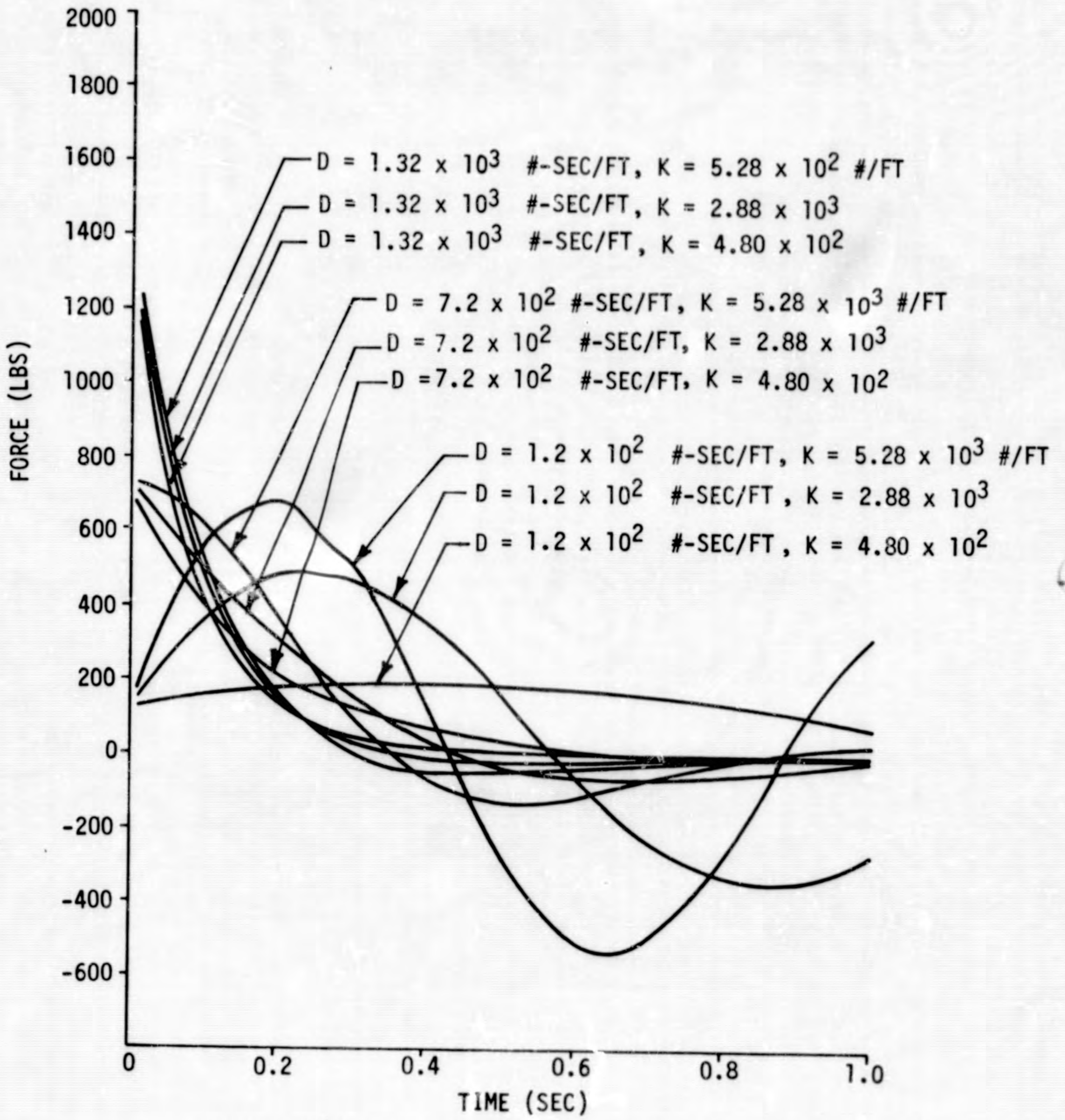


Figure 8.5-4 Tug Docking With 4160 Lb Payload

M_{TUG} = 7.124 x 10^2 Slug, $M_{Payload}$ = 1.292 x 10^2 Slug, V_o = 1.0 Ft/Sec

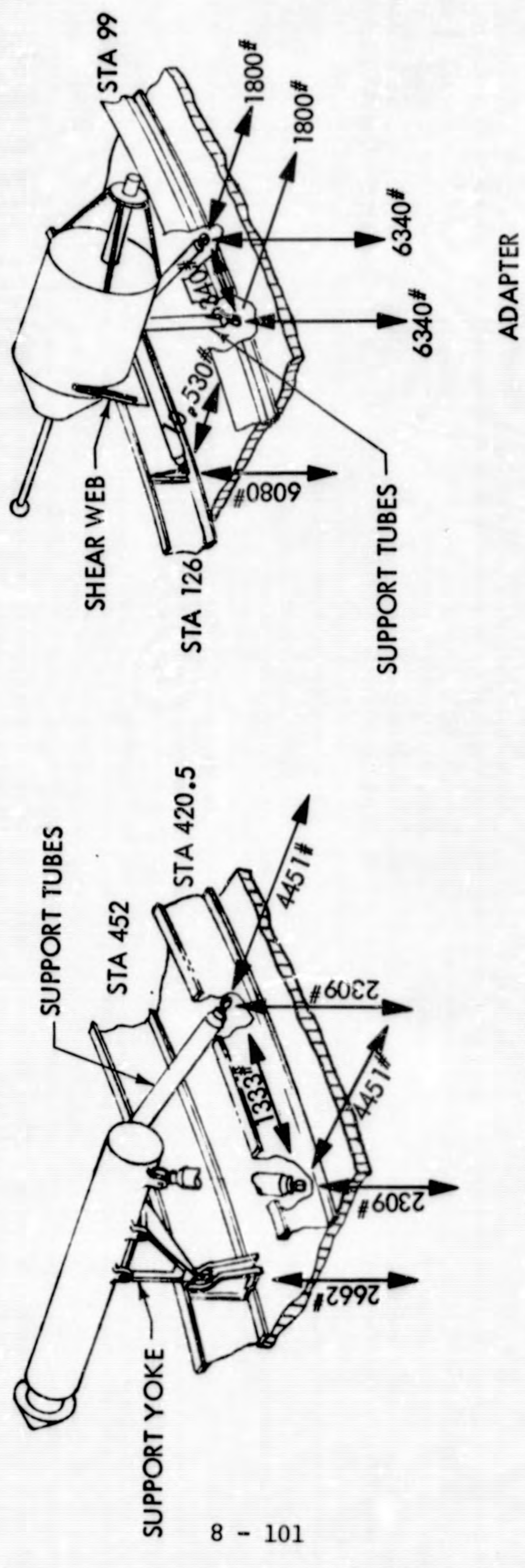


Figure 8.5-5 Docking Probe Support Loads (Ultimate)

loads of 6300 pounds axially (longitudinally and 2100 pounds laterally (Reference Page 8-97). Design reactions at the supports are shown in Figure 8.5-5. The graphite epoxy tubes are two inches in diameter, with the front tube 0.036 inches thick and the rear tubes 0.039 inches thick. The shear web is made of 0.063 inches thick aluminum sheet.

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9.0 MASS PROPERTIES

This section presents the final mass characteristics of the vehicle configurations for the three specified payloads in terms of weight, center of gravity, mass moments of inertia and products of inertia. Data are presented on an individual basis for the payload missions including the launch configuration while in the Earth Orbital Shuttle (EOS) cargo bay.

The weight data shown are the result of extensive analysis in each of the represented areas. Structural elements reflect nominal gages established from stress analyses and include the effects of manufacturing tolerances. Astrionics weights are based on supplier data for specified equipment and the associated wiring was calculated from harness layouts. A similar procedure was followed for the propulsion system equipment. Although the engine weight has been provided by the MSFC study guidelines, all other weights result from detailed analyses.

Section 9.1 presents summary weight data of each mission configuration.

Section 9.2 provides a weight breakdown of the stage at the major component/subsystem level. A more detailed weight breakdown at the structure subassembly and system component level is provided in Section 9.3.

Section 9.4 presents overall mass properties, consisting of weight, center of gravity and inertia parameters, for the dry stage and each mission launch configuration.

Section 9.5 consist of overall tug mass properties for each significant flight event starting with launch through Tug/EOS docking and reentry. Graphs are also included which present mass properties versus flight time.

Section 9.6 presents the weight effect of designing the outer shell to (1) 1972 graphite composite allowances, (2) an all-aluminum 7075-T6 material.

The alternate mission payload weights used in the derivation of mass properties have been increased to reflect Tug performance margins. The 9900 pound and 5200 pound payloads include performance margins of 1884 pounds and 1040 pounds respectively. However, the structural analysis data was based on the payload weights described in the Tug Point Design Ground Rules.

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9.1.1 SUMMARY WEIGHT STATEMENT

CONFIGURATION: 3720 LB PAYLOAD TO GEOSYNCHRONOUS ORBIT AND RETURN

STRUCTURE	1769	
THERMAL CONTROL	275	
ASTRIONICS	869	
PROPULSION	924	
DRY WEIGHT		3837
CONTINGENCY	383	
DRY WEIGHT + CONTINGENCY		4220
NONUSABLE FLUIDS	1068	
BURNOUT WEIGHT		5288
EXPANDABLE FLUIDS	54541	
GROSS TUG WEIGHT AT TUG/EOS SEPARATION		59829
PRE-IGNITION LOSSES	0	
TUG/EOS INTERFACE PROVISIONS	1451	
PAYLOAD	3720	
GROSS TUG WEIGHT AT EOS LIFTOFF		65000

9.1.2 SUMMARY WEIGHT STATEMENT

CONFIGURATION: 5200 LB PAYLOAD RETRIEVAL FROM GEOSYNCHRONOUS ORBIT

STRUCTURE	1769	
THERMAL CONTROL	275	
ASTRIONICS	869	
PROPULSION	924	
DRY WEIGHT		3837
CONTINGENCY	383	
DRY WEIGHT + CONTINGENCY		4220
NONUSABLE FLUIDS	1068	
BURNOUT WEIGHT		5288
EXPANDABLE FLUIDS	55326	
GROSS TUG WEIGHT AT TUG/EOS SEPARATION		60614
PRE-IGNITION LOSSES	0	
TUG/EOS INTERFACE PROVISIONS	1451	
PAYLOAD (NOT IN LIFT OFF WEIGHT)	0	
GROSS TUG WEIGHT AT EOS LIFTOFF		62065

9.1.3 SUMMARY WEIGHT STATEMENT

CONFIGURATION: 9900 LB PAYLOAD TO GEOSYNCHRONOUS ORBIT

STRUCTURE	1769	
THERMAL CONTROL	275	
ASTRIONICS	869	
PROPULSION	924	
DRY WEIGHT		3837
CONTINGENCY	383	
DRY WEIGHT + CONTINGENCY		4220
NONUSABLE FLUIDS	1068	
BURNOUT WEIGHT		5288
EXPANDABLE FLUIDS	48361	
GROSS TUG WEIGHT AT TUG/EOS SEPARATION		53649
PRE-IGNITION LOSSES	0	
TUG/EOS INTERFACE PROVISIONS	1451	
PAYLOAD	9900	
GROSS TUG WEIGHT AT EOS LIFTOFF		65000

9.2.1 GROUP WEIGHT STATEMENT

CONFIGURATION: STAGE DRY WEIGHT	PAGE	OF
STRUCTURE		1769
FUEL TANK (INCL SCREENS, BAFFLES, ETC.)	373	
OXIDIZER TANK (INCL SCREENS, BAFFLES, ETC.)	319	
FUEL TANK SUPPORTS	16	
OXIDIZER TANK SUPPORTS	28	
OUTER SHELL	742	
METEOROID SHIELD	46	
THRUST STRUCTURE	30	
TUG/PAYLOAD DOCKING MECH	70	
TUG/EOS DOCKING MECH.	30	
UMBILICALS	15	
SUBSYSTEM MOUNTING HARDWARE	100	
THERMAL CONTROL SYSTEM		275
FUEL TANK INSULATION	83	
OXIDIZER TANK INSULATION	60	
PURGE BAG, VALVES AND LINES	79	
THERMAL CONTROL SYSTEM	53	
ASTRIONICS		869
DATA MANAGEMENT	120	
GUIDANCE, NAVIGATION AND CONTROL	133	
COMMUNICATIONS	55	
INSTRUMENTATION	122	
ELECTRICAL POWER	166	
POWER CONVERSION AND DISTRIBUTION	198	
RENDEZVOUS AND DOCKING	75	
PROPULSION		924
MAIN ENGINE	230	
PRESSURIZATION SYSTEM	59	
FEED, FILL, DRAIN AND VENT SYSTEMS	200	
GIMBAL AND ACTUATION SYSTEM	40	
ATTITUDE CONTROL SYSTEM (THRUSTERS, LINES, VALVES, TANKS)	360	
PROPELLANT UTILIZATION SYSTEM	35	
SUBTOTAL (DRY WEIGHT)		3837

9.2.2 GROUP WEIGHT STATEMENT

CONFIGURATION	3720 LB PAYLOAD TO GEOSYNCHRONOUS ORBIT AND RETURN	PAGE	OF

DRY WEIGHT (BROUGHT FORWARD)		3837
CONTINGENCY 10% DRY WEIGHT	383	
DRY WEIGHT + CONTINGENCY		4220
NONUSABLE FLUIDS		1068
TRAPPED PROPELLANTS AND GASSES	295	
PRESSURANT	423	
MAIN ENGINE RESERVES (2% of ΔV)	350	
THERMAL CONTROL FLUIDS		
SUBTOTAL (BURNOUT WEIGHT)		5288
INFLIGHT LOSSES		54541
USABLE MAIN ENGINE PROPELLANT	54027	
USABLE ACS PROPELLANT	153	
START/STOP LOSSES	85	
VENTED PROPELLANTS	151	
FUEL CELL REACTANTS	125	
CHILLDOWN LOSSES		
SUBTOTAL GROSS TUG WEIGHT AT TUG/EOS SEPARATION		59829
PRE-IGNITION LOSSES		0
TUG/EOS INTERFACE PROVISIONS		1451
FORWARD SUPPORT RING AND DOCKING MECH	492	
PAYLOAD AND TUG SNUBBER AND DAMPER SYSTEMS	100	
PURGE SYSTEMS (INCLUDING TANKS, LINES, FLUIDS)	794	
TUG/EOS SERVICE SYSTEMS	65	
PAYLOAD WEIGHT		3720
TOTAL GROSS VEHICLE WEIGHT AT EOS LIFTOFF		65000

9.2.3 GROUP WEIGHT STATEMENT

CONFIGURATION 5200 LB PAYLOAD RETRIEVAL FROM GEOSYNCHRONOUS ORBIT		PAGE OF
DRY WEIGHT (BROUGHT FORWARD)		3837
CONTINGENCY 10% DRY WEIGHT	383	
DRY WEIGHT + CONTINGENCY		4220
NONUSABLE FLUIDS		1068
TRAPPED PROPELLANTS AND GASSES	295	
PRESSURANT	423	
MAIN ENGINE RESERVES (2% of ΔV)	350	
THERMAL CONTROL FLUIDS		
SUBTOTAL (BURNOUT WEIGHT)		5288
INFLIGHT LOSSES		55326
USABLE MAIN ENGINE PROPELLANT	54812	
USABLE ACS PROPELLANT	153	
START/STOP LOSSES	85	
VENTED PROPELLANTS	151	
FUEL CELL REACTANTS	125	
CHILLDOWN LOSSES		
SUBTOTAL GROSS TUG WEIGHT AT TUG/EOS SEPARATION		60614
PRE-IGNITION LOSSES		0
TUG/EOS INTERFACE PROVISIONS		1451
FORWARD SUPPORT RING AND DOCKING MECH	492	
PAYLOAD AND TUG SNUBBER AND DAMPER SYSTEMS	100	
PURGE SYSTEMS (INCLUDING TANKS, LINES, FLUIDS)	794	
TUG/EOS SERVICE SYSTEMS	65	
PAYLOAD WEIGHT		-
TOTAL GROSS VEHICLE WEIGHT AT EOS LIFTOFF		62065

9.2.4 GROUP WEIGHT STATEMENT

CONFIGURATION 9900 LB PAYLOAD TO GEOSYNCHRONOUS ORBIT		PAGE OF
DRY WEIGHT (BROUGHT FORWARD)		3837
CONTINGENCY 10% DRY WEIGHT	383	
DRY WEIGHT + CONTINGENCY		4220
NONUSABLE FLUIDS		1068
TRAPPED PROPELLANTS AND GASSES	295	
PRESSURANT	423	
MAIN ENGINE RESERVES (2% of ΔV)	350	
THERMAL CONTROL FLUIDS		
SUBTOTAL (BURNOUT WEIGHT)		5288
INFLIGHT LOSSES		48361
USABLE MAIN ENGINE PROPELLANT	47847	
USABLE ACS PROPELLANT	153	
START/STOP LOSSES	85	
VENTED PROPELLANTS	151	
FUEL CELL REACTANTS	125	
CHILLDOWN LOSSES		
SUBTOTAL GROSS TUG WEIGHT AT TUG/EOS SEPARATION		53649
PRE-IGNITION LOSSES		0
TUG/EOS INTERFACE PROVISIONS		1451
FORWARD SUPPORT RING AND DOCKING MECH	492	
PAYLOAD AND TUG SNUBBER AND DAMPER SYSTEMS	100	
PURGE SYSTEMS (INCLUDING TANKS, LINES. FLUIDS)	794	
TUG/EOS SERVICE SYSTEMS	65	
PAYLOAD WEIGHT		9900
TOTAL GROSS VEHICLE WEIGHT AT EOS LIFTOFF		65000

9.3 DETAIL WEIGHT STATEMENT

STRUCTURE GROUP
PRIMARY STRUCTURE

	LH_2 TANK	LOX TANK	THRUST STRUCTURE	OUTER SHELL		
				FWD	CENTER	AFT
SKIN	99		5	164	188	45
CLOSE OUT						
FRAMES						
STABILITY		5	4	18	29	
FIELD JOINT & TANK SUPT (2)					78	
DOCKING				105		33
PAYLOAD KICK FRAME				54		
STRUTS			13			
BULKHEAD - FORWARD	116	169				
BULKHEAD - AFT	116	116				
THRUST BLOCK			5			
INNER TANK - APS	18	9				
ANTIVORTEX	24	20				
TANK SUPPORTS	16	28				
ATTACH HARDWARE			3	6		22
TOTAL - SECTION	389	347	30	347	295	100
TOTAL- OUTER SHELL				742		
TOTAL - PRIMARY STRUCTURE						1508

9.3 DETAIL WEIGHT STATEMENT (Cont)

STRUCTURE GROUP
SECONDARY STRUCTURE

	METEOROID SHIELD	DOCKING TUG-PAYLOAD	DOCKING TUG-EOS	UMBILICALS	SUB-SYSTEM MOUNTS
METEOROID SHIELD					
FRONTAL MEMBRANE	20				
AFT SECTION PANELS	24				
ATTACH HARDWARE	2				
DOCKING MECHANISM					
DROGUES		39	15		
LATCHING DEVICES		24	14		
SUPPORTS		5			
ATTACH HARDWARE		2	1		
UMBILICALS				15	
SUBSYSTEM MOUNTING					100
TOTAL - SECONDARY STRUCTURE	46	70	30	15	100
TOTAL - PRIMARY STRUCTURE - BROUGHT FORWARD					1508
TOTAL - STRUCTURE					1769

9.3 DETAIL WEIGHT STATEMENT (Cont)

THERMAL CONTROL

FUEL TANK INSULATION (753 sq ft)	(83)
MLI (30 LAYERS)	41
HARDSPOTS AND POST	4
INNER SUPPORT	19
OUTER SUPPORT	19
OXIDIZER TANK INSULATION (398 sq ft)	(60)
MLI (45 LAYERS)	33
HARDSPOTS AND POST	7
INNER SUPPORT	10
OUTER SUPPORT	10
INSULATION PURGE SYSTEM	(79)
BAG/SEALER	32
VALVES	11
PLUMBING	35
HARDWARE	1
ENVIRONMENTAL CONTROL SYSTEM	(23)
PANELS - EQUIPMENT MOUNTING	25
LOUVRES	
HEATERS	
HARDWARE	
ACPS PLUME PROTECTION	(30)
INSULATION	6
EROSION BARRIER	20
HARDWARE	4
TOTAL - THERMAL CONTROL SYSTEM	275

9.3 DETAIL WEIGHT STATEMENT (Cont)

ASTRIONICS

	UNITS	ANTENNAS	CIRCUITRY	TOTAL
DATA MANAGEMENT	(120)			(120)
MAIN COMPUTER	26			26
MEASUREMENT PROCESSOR	22			22
DATA ACQUISITION UNIT	67			67
TAPE RECORDER	5			5
WIRING AND CONNECTORS				
GUIDANCE-NAVIGATION AND CONTROL	(133)			(133)
STAR TRACKER	28			28
HORIZON SENSOR	37			37
HORIZON SENSOR ELECTRONICS	14			14
BACKUP STABILIZATION ASSEM.	6			6
IMU	21			21
AUTOCOLLIMATOR	11			11
MPS & ACPS DRIVERS	16			16
WIRING AND CONNECTORS				
COMMUNICATIONS	(50)	(5)		(55)
PM TRANSPONDER	24			24
FM TRANSMITTER	2			2
COMMAND DECODER	1.5			1.5
POWER AMPLIFIER	19			19
BI-PHASE MODULATOR	1			1
ISOLATION FILTER	0.5			0.5
RF SWITCH	0.5			0.5
RF MULTIPLEXER	0.5			0.5
HYBRID JUNCTION	1			1
OMNI ANTENNA		5		5
WIRING AND CONNECTORS				
TOTAL - SUB ITEMS TO BE BROUGHT FWD	303	5		
TOTAL - TO BE BROUGHT FORWARD				308

9.3 DETAIL WEIGHT STATEMENT (Cont)

ASTRIONICS (CONT'D)

	UNITS	ANTENNAS	CIRCUITRY	TOTAL
INSTRUMENTATION	(122)			(122)
SIGNAL CONDITIONERS	51			51
SENSORS	68			68
POWER SUPPLY	3			3
WIRING				
ELECTRICAL POWER	(166)			(166)
FUEL CELL	37			37
BATTERY	12			12
CONTROLLERS	60			60
THERMAL CONTROL	43			43
PLUMBING	14			14
POWER CONVERSION & DISTRIBUTION			(198)	(198)
CIRCUIT BREAKERS				
WIRING AND CONNECTORS			198	198
RENDEZVOUS AND DOCKING	(73)	(2)		(75)
LASER RADAR	45			45
TV CAMERA	8			8
TV GIMBAL MOUNT	3			3
CAMERA LIGHTS	6			6
ANTENNA		2		2
TRANSPONDER	11			11
WIRING AND CONNECTORS				
TOTAL - SUB ITEMS THIS PAGE	361	2	198	561
TOTAL - SUB ITEMS BROUGHT FORWARD	303	5		308
TOTAL - ASTRIONICS				869

9.3 DETAIL WEIGHT STATEMENT (Cont)

PROPULSION
ENGINE INSTALLATION

ENGINE (AS SUPPLIED)	213
ENGINE SERVICE	17
BOTTLES	
PLUMBING	
WIRING AND CONNECTORS	
GIMBAL AND ACTUATION SYSTEM	(40)
PUMP/MOTOR ACTUATORS ACCUMULATOR/RESERVOIR	37
PLUMBING	1
INSTALLATION HARDWARE	1
FLUID	1
TOTAL - TO BE BROUGHT FORWARD	270

9.3 DETAIL WEIGHT STATEMENT (Cont)

PROPULSION (CONT'D)
PROPELLANT SYSTEM

	FUEL	OXIDIZER
PRESSURIZATION	(29)	(30)
REGULATOR	7	7
VALVES	4	4
DIFFUSER	2	2
PLUMBING	14	15
CLIPS AND HARDWARE	2	2
FILL & DRAIN	(16)	(16)
VALVE	7	7
DISCONNECT	4	4
LINE	2	2
PRESSURE SWITCH	2	2
INSTALLATION HARDWARE	1	1
FEED SYSTEM	(21)	(14)
VALVE	9	9
LINE	11	4
INSTALLATION HARDWARE	1	1
VENT & PURGE SYSTEM	(67)	(66)
VALVE	48	48
DISCONNECT	4	4
LINE	2	2
NON PROPULSIVE VENT	5	5
HELIUM PURGE SYSTEM	6	6
INSTALLATION HARDWARE	2	1
PROPELLANT MANAGEMENT	(23)	(12)
STILLWELL	6	3
SENSORS	2	1
PROBES		
SUPPORTS AND HARDWARE	12	6
ELECTRICAL CIRCUITRY	3	2
TOTAL - PROPELLANT SYSTEMS	156	138
TOTAL - BROUGHT FORWARD		270
TOTAL - TO BE BROUGHT FORWARD		564

9.3 DETAIL WEIGHT STATEMENT (Cont)

PROPULSION (CONT'D)
ATTITUDE CONTROL PROPULSION SYSTEM

THRUSTER		
NOZZLES (NO REQ. 14)		96
INSTALLATION HARDWARE		28
SUB-TOTAL - THRUSTERS		(124)
	FUEL	OXIDIZER
PROPELLANT SYSTEM		
TURBO PUMP	8	15
GAS GENERATOR	5	5
HEAT EXCHANGER	14	15
ACCUMULATOR	20	19
REGULATORS	14	14
VALVES	30	29
SWITCHES AND CONTROLS	9	9
PLUMBING AND FITTINS	5	5
INSTALLATION HARDWARE	10	10
SUBTOTAL - PROPELLANT SYSTEM	115	121
TOTAL - ATTITUDE CONTROL PROPULSION SYSTEM		360
TOTAL - BROUGHT FORWARD		564
TOTAL - PROPULSION		924

9.3 DETAIL WEIGHT STATEMENT (Cont)

NON USABLE PROPELLANTS
RESIDUAL AND RESERVE

	RESERVE ΔV	PRESSURANTS	TRAPPED	TOTAL
FUEL TANK	50	187	17	(254)
OXIDIZER TANK	300	236	240	(776)
MAIN PROPULSION SYSTEM			(15)	(15)
ENGINE			(5)	
OXIDIZER			2	
FUEL			3	
LINES			(10)	
OXIDIZER			9	
FUEL			1	
ATTITUDE CONTROL PROPULSION SYS.			(23)	(23)
ACCUMULATOR - OXIDIZER			18	
- FUEL			5	
SUBTOTAL	350	423	295	
TOTAL - NON USABLE PROPELLANTS				1068

9.3 DETAIL WEIGHT STATEMENT (Cont)

EXPENDABLE PROPELLANTS

	FUEL	OXIDIZER	TOTAL
USABLE PROPELLANT	(7786)	(46241)	(54027)
MAIN STAGE	7579	45473	53052
ACPS-ON ORBIT MANEUVER	207	768	975
ACPS - ATTITUDE CONTROL	(32)	(121)	(153)
START/STOP LOSSES	(41)	(44)	(85)
START	18		18
STOP	23	44	67
VENTED PROPELLANTS	(99)	(52)	(151)
PRECONDITIONING	(36)	(32)	68
ACPS	33	27	
FUEL CELL	3	5	
THERMAL DYNAMIC VENTING			
BOIL OFF	63	20	83
FUEL CELL REACTANTS	(14)	(111)	(125)
CHILL DOWN LOSSES			
SUBTOTAL -	7972	46569	
TOTAL - EXPENDABLE PROPELLANTS			54541

9.3 DETAIL WEIGHT STATEMENT (Cont)

TUG CHARGEABLE INTERFACE PROVISIONS

TUG/EOS INTERSTAGE	(492)
STRUCTURE	(402)
SKIN	250
CLOSEOUT	
FRAME	
STABILITY	16
DOCKING/SEPARATION	33
ATTACH.	103
DOCKING MECHANISM	(90)
DROGUES INSTALLATION	
LATCHING SYSTEM	
SNUBBER AND DAMPER SYSTEM	(100)
PURGE SYSTEM	(794)
CONTAINERS	612
VALVES AND REGULATORS	15
PLUMBING	15
SUPPORTS	80
HELIUM	72
EOS/TUG SERVICE SYSTEMS	(65)
TOTAL - TUG CHARGEABLE INTERFACE PROVISIONS	1451

9.4.1 SYSTEMS MASS PROPERTIES

CONFIGURATION: 3720 PAYLOAD TO GEOSYNCHRONOUS ORBIT AND RETURN PAGE OF

SYSTEM	WEIGHT LB	CENTER OF GRAVITY INCHES			MOMENT OF INERTIA SLUG FT^2		
		X	Y	Z	I_{x-x}	I_{y-y}	I_{z-z}
STRUCTURE	1769	279.6	0.3	0.6	2175	5850	5839
THERMAL CONTROL	275	273.1	2.5	-2.2	299	754	752
ASTRIONICS	869	298.9	9.9	2.3	971	3898	3851
PROPULSION	924	159.4	1.2	17.3	525	2540	2392
CONTINGENCY	383	254.0	2.8	4.8	360	1761	1761
DRY STAGE	4220	254.5	2.8	4.7	4387	17280	17044
NON USABLE FLUIDS	718	201.6	-0.5	1.3	27	1167	1150
RESERVE FLUIDS	350	147.4	0.0	0.0	0	108	108
BURNOUT WEIGHT	5288	240.2	2.2	3.9	4418	19627	19372
MAIN ENGINE PROPELLANT	54027	201.3	0.0	0.0	--	37515	37515
ACPS PROPELLANT	153	211.8	0.0	0.0	--	143	143
START/STOP LOSSES	85	255.4	0.0	0.0	--	119	119
VENTED PROPELLANT	151	283.9	0.0	0.0	--	192	192
FUEL CELL REACTANT	125	196.1	0.0	0.0	--	70	70
GROSS TUG WEIGHT AT TUG/EOS SEPARATION	59829	205.0	0.2	0.3	4438	59488	59129
FORWARD SUPPORT RING	492	80.3	0.0	0.0	689	434	434
SNUBBER AND DAMPER SYSTEM	100	166.0	0.0	0.0	187	311	311
PURGE SYSTEM	794	50.0	0.0	0.0	617	302	302
EOS/TUG SERVICE SYSTEMS	65	240.0	0.0	0.0	118	176	176
PAYLOAD	3720	605.0	0.0	0.0	3245	7658	7658
TOTAL GROSS WEIGHT	65000	223.8	0.2	0.3	8728	227406	227144

9.4.2 SYSTEMS MASS PROPERTIES

CONFIGURATION: 5200 LB PAYLOAD RETRIEVED FROM GEOSYNCHRONOUS ORBIT					PAGE	OF	
		CENTER OF GRAVITY			MOMENT OF INERTIA		
		INCHES			SLUG FT^2		
SYSTEM	WEIGHT LB	X	Y	Z	I_{x-x}	I_{y-y}	I_{z-z}
STRUCTURE	1769	279.6	0.3	0.6	2175	5850	5839
THERMAL CONTROL SYSTEM	275	273.1	2.5	-2.2	299	754	752
ASTRIONICS	865	298.9	9.9	2.3	971	3898	3851
PROPULSION	924	159.4	1.2	17.3	525	2540	2392
CONTINGENCY	383	254.0	2.8	4.8	360	1761	1761
DRY STAGE	4220	254.5	2.8	4.7	4387	17280	17044
NON USABLE FLUIDS	718	201.6	-0.5	1.3	27	1167	1150
RESERVE FLUIDS	350	147.4	0.0	0.0	0	108	108
BURNOUT WEIGHT	5288	240.2	2.2	3.9	4418	19627	19372
MAIN ENGINE PROPELLANT	54812	199.4	0.0	0.0		38000	38000
ACPS PROPELLANT	153	211.8	0.0	0.0	--	143	143
START/STOP LOSSES	85	255.4	0.0	0.0	--	119	119
VENTED PROPELLANT	151	283.9	0.0	0.0	--	192	192
FUEL CELL REACTANT	125	196.1	0.0	0.0	--	70	70
GROSS TUG WEIGHT AT TUG/EOS SEPARATION	60614	203.3	0.2	0.3	4418	87805	87542
FORWARD SUPPORT RING	492	80.3	0.0	0.0	689	434	434
SNUBBER AND DAMPER SYSTEM	100	166.0	0.0	0.0	187	311	311
PURGE SYSTEM	794	50.0	0.0	0.0	617	302	302
EOS/TUG SERVICE SYSTEMS	65	240.0	0.0	0.0	118	176	176
PAYLOAD	-						
TOTAL GROSS WEIGHT	62065	200.3	0.2	0.3	5481	94610	94348

9.4.3 SYSTEMS MASS PROPERTIES

CONFIGURATION: 9900 LB PAYLOAD TO GEOSYNCHRONOUS ORBIT					PAGE OF		
SYSTEM	WEIGHT LB	CENTER OF GRAVITY INCHES			MOMENT OF INERTIA SLUG FT^2		
		X	Y	Z	I_{x-x}	I_{y-y}	I_{z-z}
STRUCTURE	1769	279.6	0.3	0.6	2175	5850	5829
THERMAL CONTROL SYSTEM	275	273.1	2.5	-2.2	299	754	752
ASTRIONICS	869	298.9	9.9	2.3	971	3898	3851
PROPULSION	924	159.4	1.2	17.3	525	2540	2392
CONTINGENCY	383	254.0	2.8	4.8	360	1761	1761
DRY STAGE	4220	254.5	2.8	4.7	4387	17280	17044
NON USABLE FLUIDS	718	201.6	-0.5	1.3	27	1167	1150
RESERVE FLUIDS	350	147.4	0.0	0.0	0	108	108
BURNOUT WEIGHT	5288	240.2	2.2	3.9	4418	19627	19372
MAIN ENGINE PROPELLANT	47847	194.3	0.0	0.0	--	37515	37515
ACPS PROPELLANT	153	211.8	0.0	0.0	--	143	143
START/STOP LOSSES	85	255.4	0.0	0.0	--	119	119
VENTED PROPELLANT	151	283.9	0.0	0.0	--	192	192
FUEL CELL REACTANT	125	196.1	0.0	0.0	--	70	70
GROSS TUG WEIGHT AT TUG/EOS SEPARATION	53649	199.2	0.2	0.4	4422	78443	78257
FORWARD SUPPORT RING	492	80.3	0.0	0.0	689	434	434
SNUBBER AND DAMPER SYSTEM	100	166.0	0.0	0.0	187	311	311
PURGE SYSTEM	794	50.0	0.0	0.0	617	302	302
EOS/TUG SERVICE SYSTEMS	65	240.0	0.0	0.0	118	176	176
PAYLOAD	9900	605.0	0.0	0.0	8636	20379	20379
TOTAL GROSS WEIGHT	65000	258.3	0.2	0.3	14123	407289	407227

9.5.1 SEQUENCE MASS PROPERTIES STATEMENT

CONFIGURATION: 3720 LB PAYLOAD TO GEOSYNCHRONOUS ORBIT AND RETURN					PAGE OF		
		CENTER OF GRAVITY INCHES			MOMENT OF INERTIA SLUG FT^2		
MISSION EVENT	WEIGHT LB	X	Y	Z	I_{x-x}	I_{y-y}	I_{z-z}
GROSS IN EOS CARGO BAY (ASCENT)	65000	223.8	0.2	0.3	8728	227406	227144
TUG SUPPORT AND INTERFACE SYS.	- 1451						
SEPARATION PROPELLANT	- 67						
IGNITION	63482	227.1	0.2	0.3	7660	218436	218174
PROPELLANT	-27287						
ORBIT INSERTION (100 X SYNC)	36195	231.7	0.3	0.5	7659	181730	181469
PROPELLANT	-11721						
CIRCULARIZED ORBIT (GEOSYNCHRONOUS)	24474	251.0	0.5	0.8	7657	163267	163006
PROPELLANT	- 198						
DEPLOY PAYLOAD	- 3720						
PHASE FOR RENDEZVOUS 2ND PAYLOAD	20556	187.1	0.6	1.0	4409	36721	36461
PROPELLANT	- 534						
PAYLOAD	+ 3720						
PHASE FOR RETURN TO SHUTTLE	23742	252.5	0.5	0.8	7657	161990	161729
PROPELLANT	- 7540						
ORBIT INSERTION (270 CIRCULAR)	16202	288.7	0.7	1.2	7655	141128	140868
PROPELLANT	- 6426						
ORBIT INSERTION (100 CIRCULAR)	9776	373.5	1.2	2.0	7650	98308	98050
PROPELLANT	- 768						
DOCK TO SHUTTLE	9008	391.0	1.3	2.2	7649	90080	89823
VENT PROPELLANT	- 1068						
PURGE GAS	+ 72						
TUG SUPPORT AND INTERFACE SYS.	+ 1379						
GROSS IN EOS CARGO BAY (RETURN)	9391	367.1	1.3	2.1	8662	109142	108899

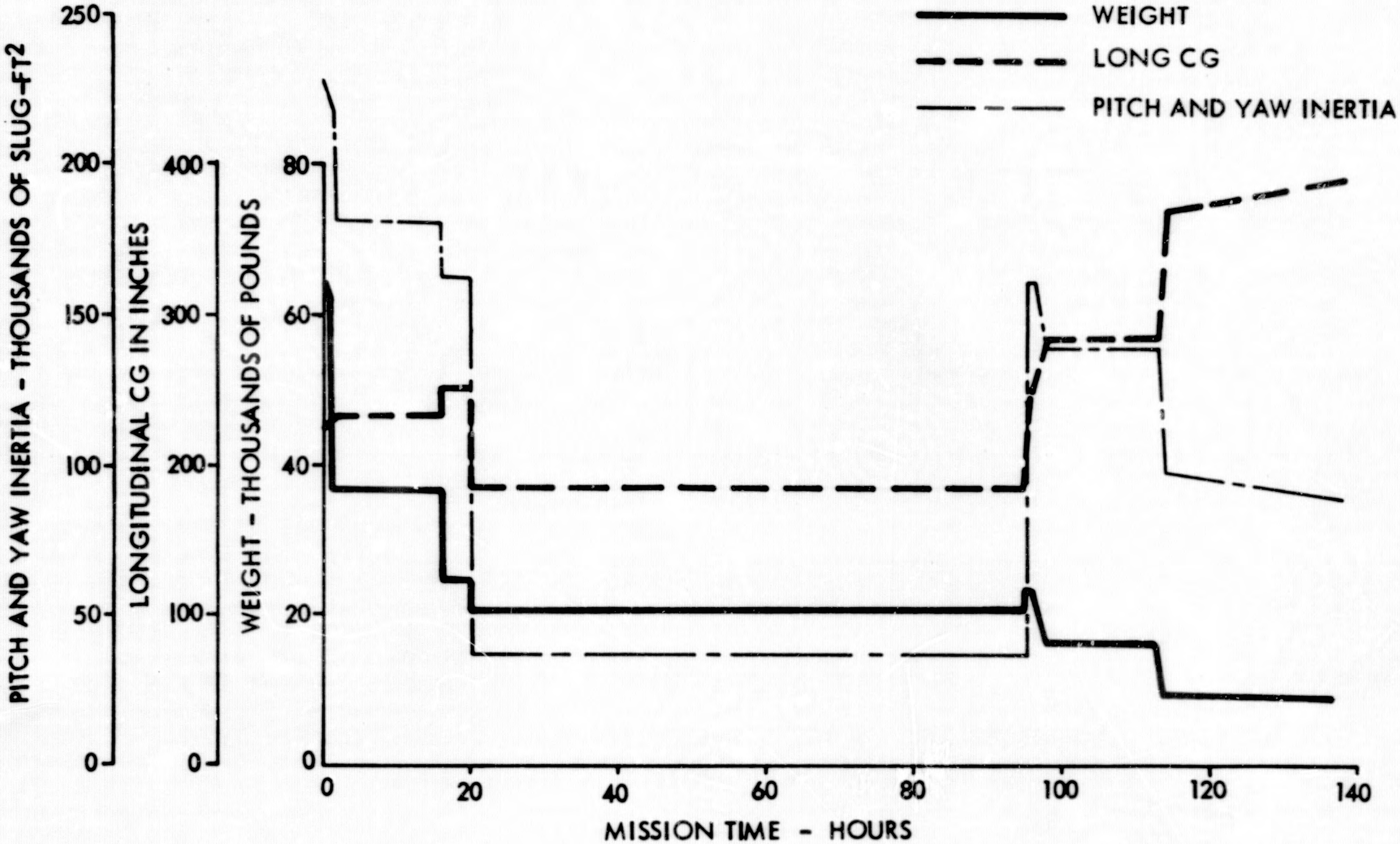

Figure 9.5-1 Mass Properties vs Mission Time 3720 Lb Payload to Geosynchronous Orbit and Return

9.5.2 SEQUENCE MASS PROPERTIES STATEMENT

CONFIGURATION: 5200 LB PAYLOAD RETRIEVED FROM GEOSYNCHRONOUS ORBIT					PAGE OF		
		CENTER OF GRAVITY			MOMENT OF INERTIA		
		INCHES			SLUG FT^2		
MISSION EVENT	WEIGHT LB	X	Y	Z	I_{x-x}	I_{y-y}	I_{z-z}
GROSS IN EOS CARGO BAY (ASCENT)	62065	200.3	0.2	0.3	5481	94610	94348
TUG SUPPORT AND INTERFACE SYS.	- 1451						
SEPARATION PROPELLANT	- 57						
IGNITION	60557	203.2	0.2	0.3	4413	87644	87382
PROPELLANT	-25976						
ORBIT INSERTION (100 X SYNC)	34581	188.9	0.3	0.6	4411	50748	50487
PROPELLANT	-12210						
CIRCULARIZED ORBIT (GEOSYNCHRONOUS)	22371	187.0	0.5	0.9	4409	38529	38268
PROPELLANT	- 165						
PHASE FOR PAYLOAD RETRIEVAL	22206	187.0	0.5	0.9	4409	38329	38068
PROPELLANT	- 138						
PAYLOAD	+ 5200						
PHASE FOR RETURN TO SHUTTLE	27268	266.4	0.4	0.7	8950	207886	207625
PROPELLANT	8736						
ORBIT INSERTION (270 CIRCULAR)	18532	307.9	0.6	1.1	8948	178427	178167
PROPELLANT	- 7334						
ORBIT INSERTION (100 CIRCULAR)	11198	404.0	1.1	1.8	8944	116583	116325
PROPELLANT	- 710						
DOCK TO SHUTTLE	10488	421.2	1.1	1.9	8943	105692	105435
VENT PROPELLANT	- 1068						
PURGE GAS	+ 72						
TUG SUPPORT AND INTERFACE SYS.	+ 1379						
GROSS IN EOS CARGO BAY (RETURN)	10871	399.4	1.1	1.8	9956	127816	127572

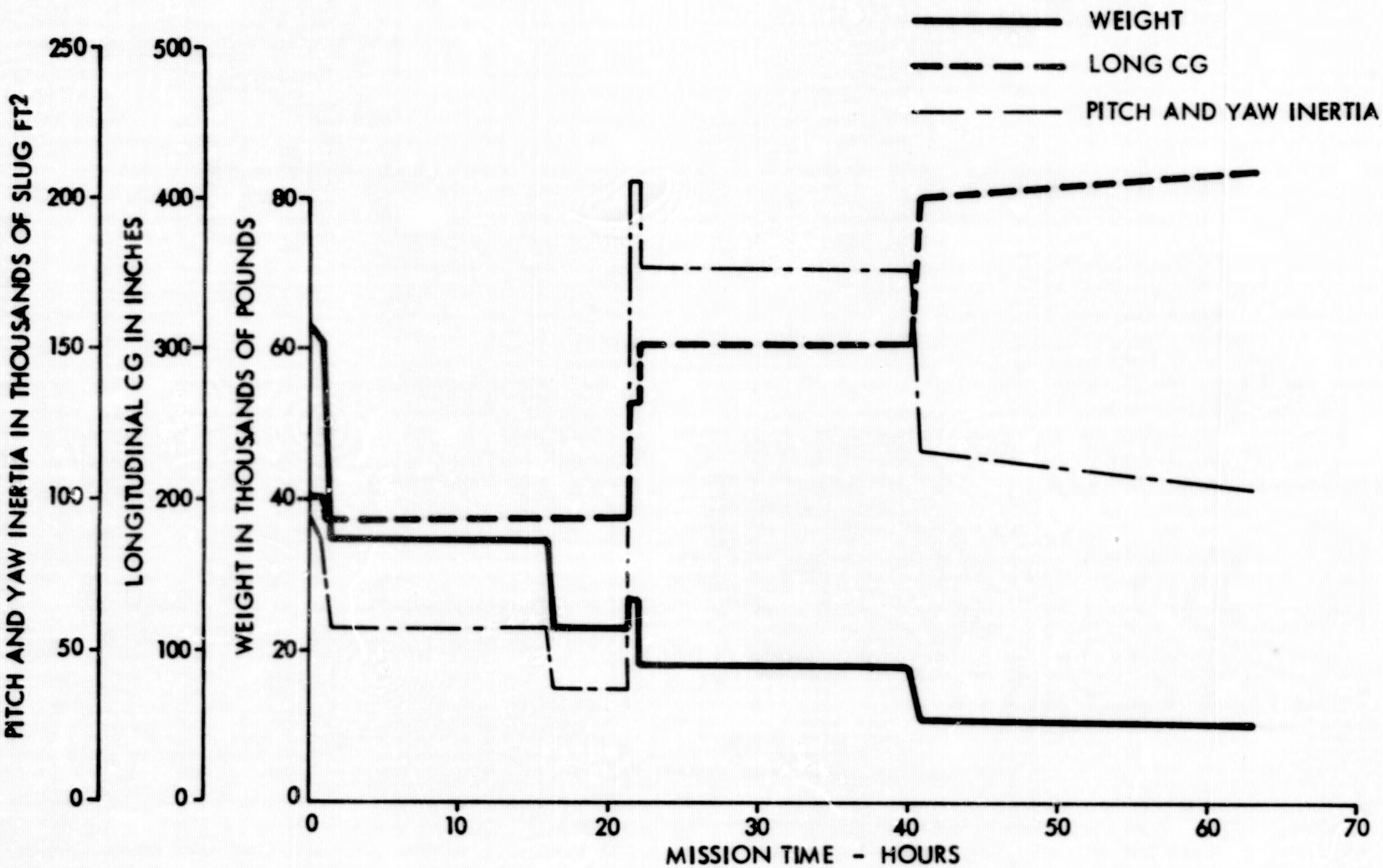

Figure 9.5-2 Mass Properties vs Mission Time 5200 Lb Payload Retrieval From Geosynchronous Orbit

9.5.3 SEQUENCE MASS PROPERTIES STATEMENT

CONFIGURATION: 9900 LB PAYLOAD TO GEOSYNCHRONOUS ORBIT					PAGE OF		
		CENTER OF GRAVITY			MOMENT OF INERTIA		
		INCHES			SLUG FT^2		
MISSION EVENT	WEIGHT LB	X	Y	Z	I_{x-x}	I_{y-y}	I_{z-z}
GROSS IN EOS CARGO BAY (ASCENT)	65000	258.3	0.2	0.3	14123	407489	407227
TUG SUPPORT AND INTERFACE SYS.	- 1451						
SEPARATION PROPELLANT	- 100						
IGNITION	63449	262.5	0.2	0.3	13055	394900	394638
PROPELLANT	-26653						
ORBIT INSERTION (100 X SYNC)	36796	299.5	0.3	0.5	13054	336332	336070
PROPELLANT	-11758						
CIRCULARIZED ORBIT (GEOSYNCHRONOUS)	25038	354.5	0.5	0.8	13053	273760	273500
PROPELLANT	- 138						
DEPLOY PAYLOAD	- 9900						
PHASE FOR RETURN TO SHUTTLE	15000	190.1	0.8	1.3	4407	31413	31155
PROPELLANT	- 4946						
ORBIT INSERTION (270 CIRCULAR)	10054	201.5	1.2	2.0	4403	26676	26418
PROPELLANT	- 4202						
ORBIT INSERTION (100 CIRCULAR)	5852	234.1	2.0	3.4	4395	20746	20493
PROPELLANT	- 564						
DOCK TO SHUTTLE	5288	240.5	2.2	3.8	4393	19786	19533
VENT PROPELLANT	- 1068						
PURGE GAS	+ 72						
TUG SUPPORT AND INTERFACE SYS.	+ 1379						
GROSS IN EOS CARGO BAY (RETURN)	5671	210.9	2.1	3.5	5406	26185	25945

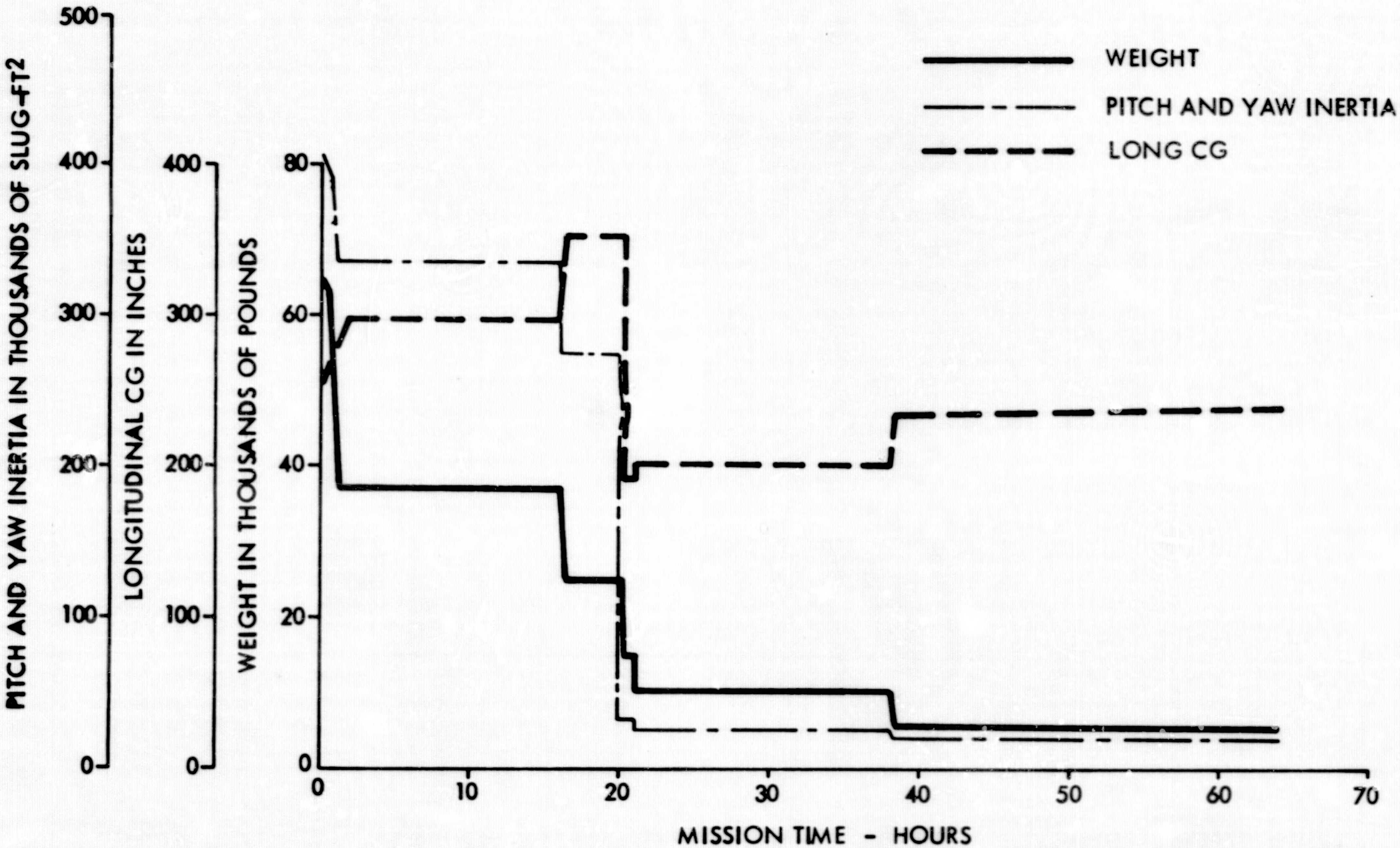

Figure 9.5-3 Mass Properties vs Mission Time 9900 Lb Payload to Geosynchronous Orbit

9.6 OUTER SHELL ALTERNATES WEIGHT COMPARISON

SECTION	AREA (SQ FT)	WEIGHT (LB)		
		POINT DESIGN	1972 GRAPHITE COMPOSITE ALLOWABLES	ALUMINUM 7075-T6
FWD SKIRT	579.5	(341)	(353)	(572)
SKINS		164	176	218
FRAMES		177	177	354
INTERTANK SHELL	555.0	(295)	(308)	(455)
SKIN		188	201	241
FRAMES		107	107	214
AFT SKIRT	99.0	(78)	(81)	(133)
SKIN		45	48	67
FRAMES		33	33	66
ADAPTER	339	(402)	(440)	(744)
SKIN		250	288	440
FRAMES		152	152	304
TOTAL - STAGE		714•	742•	1160•
- ADAPTER		402••	440••	744••
MASS FRACTION (λ)		0.903	0.902	0.896

•DOES NOT INCLUDE ATTACHING HARDWARE (28 LB)
••DOES NOT INCLUDE DOCKING PROVISIONS (90 LB)

10.0 GROUND SUPPORT EQUIPMENT

10.1 MECHANICAL

10.1.1 Parameters

- Mechanical GSE for the Tug Vehicle will be provided to support the following operational and test areas: Post Manufacturing Checkout and Acceptance Testing; Static Firing; Thermal Vacuum Testing; Maintenance level I Diagnostic Checkout and Simulated Flight Tests; Tug-to-Payload Mating Verification and Integrated System Tests; Launch Preparation and Launch; Safing Area; Alternate Landing Sites; and Storage Operations. The GSE will be designed to meet the specific site requirements. The Tug Transporter and equipment required during transportation and at the alternate landing sites will be designed to meet air mobility requirements.

10.1.2 Ground Rules

- Mechanical GSE provisioning is based on the manufacture and support of 10 vehicles making 20 flights each over a period of 10 years.

- A limited number of vehicles will be subjected to static firing and thermal vacuum testing. (GSE will be minimized in these areas and maximum utilization of existing test facility equipment is planned).

- To minimize handling of the Tug, checkout and maintenance operations subsequent to post manufacturing checkout, will be conducted on the Tug in a horizontal position in the Tug Transporter.

- Contingency payload removal at the launch pad will depend upon the payload removal system provided for the Shuttle. (Undefined at this time). Mechanical GSE provided will consist of the necessary adaptors to adapt the Tug vehicle to the basic removal system.

- GSE to effect Tug-Payload mating and transportation in a mated condition will be provided under separate contract and will be considered GFP.

- To the fullest extent possible modification of existing GSE will be utilized to support Tug requirements.

- GSE will not be provided for the maintenance level II and III areas except that associated with handling equipment on or off the Tug from the level I maintenance area.

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10.1.3 Concept

- The equipment categories of mechanical GSE to support the Tug vehicle include the normal servicing, checkout (c/o), auxiliary and handling equipment. For GSE functional areas of operation see Figure 10.1-1. Post manufacturing acceptance checkout will be accomplished in a vertical position on a facility base mount. Access will be provided by facility work platforms. Simulated flight and automatic checkout acceptance testing will be supported by Tug GSE. Subsequent to post mfg. checkout the Tug will be positioned into its transporter and transferred to the static firing site. Static firing servicing requirements and pneumatic checkout will be supported by the static firing site equipment. Operations unique to the Tug will be supported by the GSE such as: Umbilical disconnect, hydraulic servicing, etc. A similar approach will be taken at the thermal vacuum testing facility where only Tug-peculiar operations will be supported by the GSE. (Static firing and thermal vacuum testing will be conducted on a limited number of vehicles). At the maintenance level I area the Tug will remain on its transporter throughout testing. A full complement of GSE will be provided to support diagnostic checkout, maintenance operations, equipment installation and removal, and simulated flight tests. Subsequent to level I maintenance the Tug will be removed from its transporter and transferred to the Tug-payload transporter.

  The Tug transporter will be recycled at this point back to the Manufacturing area for use with a new vehicle, or to the recovery landing site for use on a returning vehicle. Handling and verification GSE will be provided for interfacing the Tug with the payload on the GFP Tug/Payload transporter. Subsequent to maintenance level I operations the Tug/Payload will be transported to the orbiter loading bay. GSE will be provided for handling, interfacing and checkout of interfaces between the orbiter and Tug. Orbiter cargo bay access equipment will be provided under the Shuttle Program. After positioning the orbiter and booster with payload on the Shuttle transporter, Tug GSE will be provided to conduct launcher-to-orbiter payload interface verification tests in preparation for launch. After Tug servicing equipment has been activated at the launch site, the Tug portable transport servicing equipment, for tank positive pressure and insulation purge, will be removed and recycled to the post manufacturing checkout area for use on the next vehicle, or to the recovery landing site for use on a returning vehicle. Launch will be supported by Tug servicing GSE located on the launcher. Existing GSE (Sat V) will be utilized, by modification, to the fullest extent possible to meet the Tug servicing requirements. For recovery operations GSE will be provided at the safing area or alternate landing site to provide for Tug safing, purging, handling and transportation. In addition to the normal manufacturing checkout, maintenance and flight operations, GSE will be provided for special Tug field operations such as tank entry and storage. GSE pallets will be provided for vertical

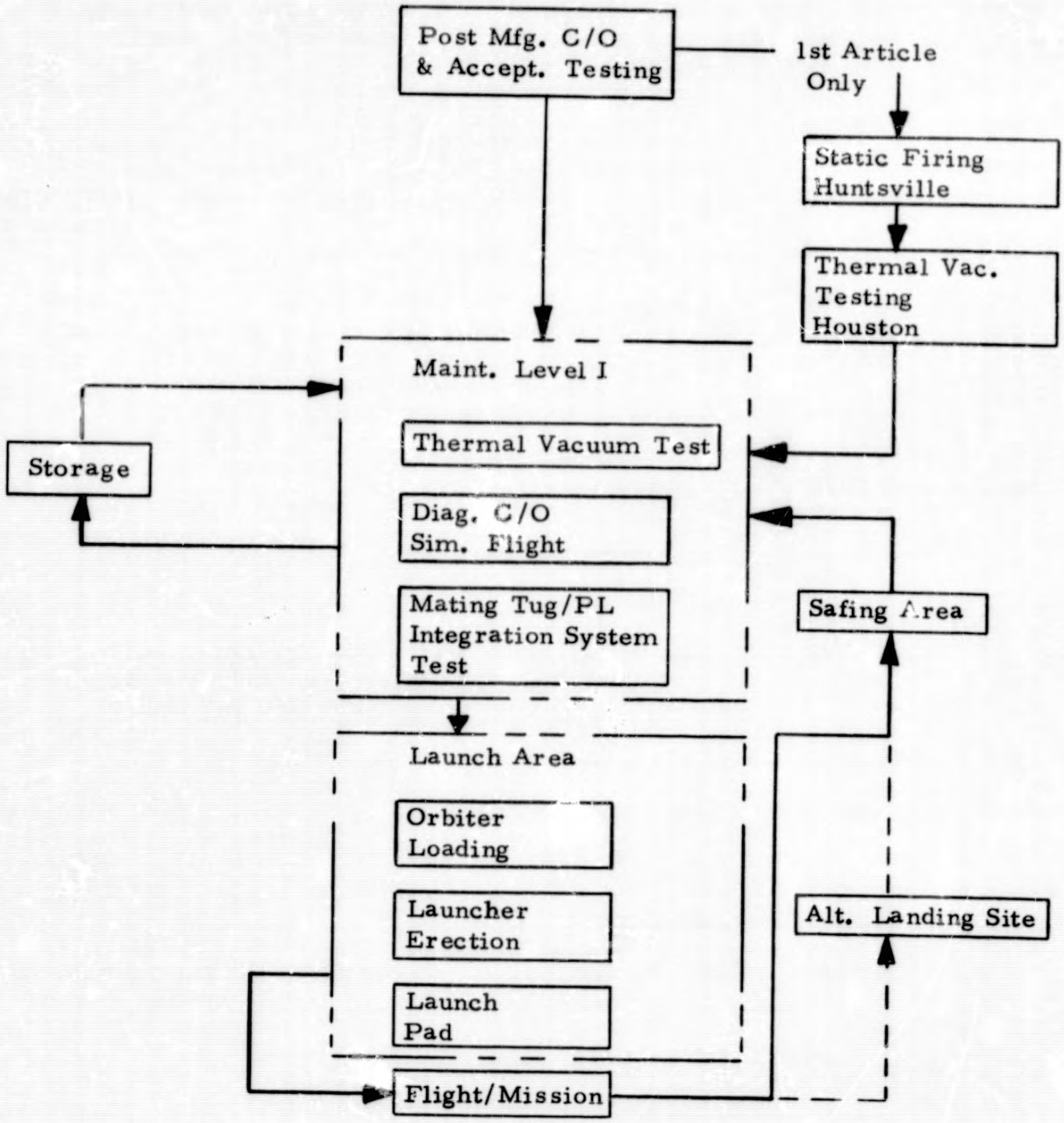

Figure 10.1-1 GSE Functional Areas and Tug Flow

storage and tank entry operations. Tank entry will require GSE access stands and environmental enclosures. Air conditioning equipment will be provided as GFP for tank and insulation environmental control.

10.1.4 Equipment

- The conceptual identification of GSE to support the checkout, servicing and handling functions of the Tug is included below. The equipment is segregated into four classes of GSE. The four classes are: Servicing, Checkout, Auxiliary and Handling Equipment.

  Each GSE item description includes the functional title, a brief description of the items purpose or function; the source, or where the equipment is derived from, (i.e., new design, modification of existing equipment, etc.); and finally the location/s of the equipment during operational usage. (Refer to Figure 10.1-1 for GSE functional areas).

10.1.5 GSE Identification List

Servicing

Title: Vacuum Pump

Purpose: Provides differential pressure across APS propellant tank and feed line propellant subcooling coil to achieve required flow of gaseous hydrogen thru the coil.

Source: New

Location: Launch Tower and Static Firing

Title: Pneumatic Servicing Console

Purpose: Distribute and control gaseous fluids to the Tug vehicle for pressurization and purge of vehicle systems preparatory for launch.

Source: New or -
Optional source - Modified Saturn V S-II, S7-41 pneumatic servicing console. (Currently recommended for Shuttle usage).

Location: Launch Tower

Title: Purging Control and Monitoring Unit

Purpose: Provides mobile, air transportable, pneumatic purge unit for safing the Tug in the safing area or at alternate landing sites. Provides electrical power, control and monitoring of Tug systems as required to perform the safing purge. Controls inert gas from facility source for purging and inerting on-board systems.

Source: New

Location: Safing Area and Alternate Landing Site

Auxiliary

Title: Blower/Environmental Control Unit

Purpose: Provides contaminant control of conditioned air at a low positive pressure for inflation and purging of LH_2 and LOX access shelters during tank entry operations.

Source: GFP

Location: Maintenance Level I and Storage

Title: Protective Support Pads - LOX Tank

Purpose: Pads will be utilized during tank entry to evenly distribute personnel weight to lower LO_2 tank wall and provide protection from damage during tank entry operations.

Source: New

Location: Maintenance Level I and Storage

Title: Protective Support Pads - LH_2 Tank

Purpose: Pads will be utilized during tank entry to evenly distribute personnel weight to lower LH_2 tank wall and provide protection from damage during tank entry operations.

Source: New

Location: Maintenance Level I and Storage

Title: Protective Cover, LH_2 Manhole

Purpose: Cover provides protection to LH_2 tank manhole sealing surfaces during tank entry and provides adaption to the environmental shelter utilized for contaminant control during tank entry.

Source: New

Location: Maintenance Level I and Storage

Title: Shelter, Environmental, Control, LH_2 Tank Access

Purpose: Shelter provides environmentally controlled work area during LH_2 tank entry or forward area barrier entry. Provides control for working atmosphere and contaminant control protection. Shelter adapts to installed forward work platform with Tug in vertical position.

Source: New

Location: Maintenance Level I and Storage

Title: Flood Light Set, Internal LH_2 Tank

Purpose: Provides internal tank lighting to support internal tank operations (LOX and LH_2).

Source: New

Location: Maintenance Level I and Storage

Title: Ladder, LH_2 Tank Entry

Purpose: Ladder will be utilized during LH_2 tank entry with Tug in vertical position. Ladder adapts to LH_2 manhole protective cover and internal tank bottom protective support bags.

Source: New

Location: Maintenance Level I and Storage

Title: Ladder, LOX Tank Entry

Purpose: Ladder will be utilized during LOX tank entry with Tug in vertical position. Ladder adapts to LOX manhole protective cover and internal tank bottom protective support bags.

Source: New

Location: Maintenance Level I and Storage

Title: Protective Cover, LOX Manhole

Purpose: Cover provides protection to LOX tank manhole sealing surfaces during tank entry operations.

Source: New

Location: Maintenance Level I and Storage

Title: Protective Cover, Inter-Tank Area Access Door

Purpose: Cover provides protection to the inter-tank access door sealing surfaces during inter-area entry and provides adaptation to the environmental shelter.

Source: New

Location: Maintenance Level I and Storage

Title: Shelter, Environmental and Contaminant Control, LOX Tank Access

Purpose: Shelter provides environmentally controlled work area access operations. Provides controlled working atmosphere and contaminant control protection. Shelter adapts to the positioned inter-area external access platform with the Tug in a vertical position.

Source: New

Location: Maintenance Level I and Storage

Title: Platform, Inter-Tank Area and LOX Tank Access, External

Purpose: Provides a work platform level in the area of the inter-tank area access door. Platform will provide external access to the inter-tank access door and is utilized in conjunction with the internal inter-tank access platforms.

Source: New

Location: Maintenance Level I and Storage

Title: Platform, Inter-Tank Area and LOX Tank Access, Internal

Purpose: Provides a work platform capable of being installed in the inter-tank area through the inter-tank access door. Platforms will provide work access to the components and lines located in the inter-tank area and access to the LOX tank manhole for tank entry.

Source: New

Location: Maintenance Level I and Storage

Title: Platform, Work and LH_2 Tank Access, Forward

Purpose: Provides a work platform providing structural protection to the LH_2 tank forward insulation area and an access platform to the LH_2 tank access manhole. Unit is designed for use during LH_2 tank entry and/or forward area maintenance with the Tug positioned vertically.

Source: New

Location: Maintenance Level I and Storage

Title: LN_2 Heat Exchanger

Purpose: Provide thermal conditioning of gases supplied to the "Tug" from the pneumatic servicing console for the APS GH_2 and GOX accumulators.

Source: Modified Saturn V S-II, A7-71 LH_2 heat exchanger. Unit to be modified to provide liquid nitrogen bath in lieu of current liquid hydrogen bath. Existing unit can satisfy Shuttle and Tug requirements with liquid nitrogen bath and common usage to both vehicles.

Location: Launch Tower and Static Firing Area

Title: Hydraulic Servicing Unit

Purpose: Provides the capability to replenish the hydraulic actuator reservoir. (Make-up supply). Unit could be used on a contingency basis with payload (Tug) installed in the orbiter. (Portable hand held unit).

Source: New

Location: Launch Tower, Maintenance Level I, Post Manufacturing Checkout and Static Firing Area

Title: Hydraulic Accumulator Charging Unit

Purpose: Provides the capability to recharge the hydraulic actuator accumulator. Unit to be used on a contingency basis with the payload (Tug) installed in the orbiter.

Source: New

Location: Launch Tower

Title: Umbilical Disconnect Assembly

Purpose: Provides retractable mounting base for fluid and electrical disconnects and connectors providing ground venting, servicing, control and monitoring. Interfaces with orbiter payload umbilical plate.

Source: New

Location: Launch Tower Payload Umbilical Retraction Arm

Title: Platform Access and Protection, Forward

Purpose: Provides access work platforms structural and personnel protection for contingency operations on the Tug while stowed vertically in the orbiter cargo bay.

Source: New

Location: Adapts to Tug in orbiter cargo bay in vertical launch position on launch tower

Title: Platform, Access and Protection, Aft

Purpose: Provides access, work platforms and personnel protection for contingency operations on the Tug while stowed vertically in the orbiter cargo bay.

Source: New

Location: Adapts to Tug in orbiter cargo bay in vertical launch positions on launch tower.

Title: Forward Environmental Work Area Shelter

Purpose: Provides environmentally controlled work area shelter when access is required thru the forward purge barrier in non-controlled areas.

Source: New

Location: Maintenance Level I or Optional Areas

Title: Aft Environmental Work Area Shelter

Purpose: Provides environmentally controlled work area shelter when access is required thru the aft purge barrier in non-controlled areas.

Source: New

Location: Maintenance Level I or Optional Areas

Title: Environmental Protection Shroud

Purpose: Provides encapsulating shroud to protect the Tug vehicle during transportation and temporary external storage.

Source: New

Location: Transportation and Optional Areas

Title: Positive Pressurization and Purge Unit

Purpose: Provides independent, portable source and control of conditioned inert gas for positive pressurization of LOX and LH_2 tank and purge of tank insulation during ground transportation and periods the Tug vehicle is not connected to C/O, maintenance or launch pneumatic servicing equipment.

Source: New

Location: Multiple areas; safing area, transportation, and Alternate site primarily.

Title: Engine Portective Cover

Purpose: Provide protective shroud around main engine thrust chamber cone to prevent damage during handling, checkout and maintenance operations.

Source: New

Location: Maintenance Level I, Transportation, Post Manufacturing Checkout, Static Firing, and Landing Areas.

Title: Umbilical Disconnect Assembly - Forward (LH_2 Panel)

Purpose: Provides mounting base for ground fluid and electrical disconnects and connectors to provide ground venting, servicing, control and monitoring during checkout and maintenance operations. Interfaces GSE with Tug forward umbilical plate.

Source: New

Location: Maintenance Level I, Post Manufacturing Checkout, Static Firing and Thermal Vacuum Testing

Title: Umbilical Disconnect Assembly, Aft (LOX Panel)

Purpose: Provides mounting base for ground fluid and electrical disconnects and connectors to provide ground venting, servicing, control and monitoring during checkout and maintenance operations. Interfaces GSE with Tug aft umbilical plate.

Source: New

Location: Maintenance Level I, Post Manufacturing Checkout, Static Firing and Thermal Vacuum Testing

Title: Access Stand, Horizontal, Aft

Purpose: Provides access to intermediate and upper levels of aft end of Tug when mounted in horizontal attitude work stand. Designed for use with aft skirt removed.

Source: New

Location: Maintenance Level I Area

Title: Access Stand, Horizontal, Forward

Purpose: Provides access to intermediate and upper levels of forward end of Tug when mounted in horizontal attitude work stand.

Source: New

Location: Maintenance Level I Area

Title: Access Stand, Horizontal, Side

Purpose: Provides access to intermediate and upper levels of Tug exterior side with Tug in horizontal attitude work stand. Supports checkout and maintenance.

Source: New

Location: Maintenance Level I Area

Title: Adapter Ring, Separation Plane

Purpose: Provides structural support to Tug aft separation plane for support in horizontal position when aft skirt is not installed for maintenance operations.

Source: New

Location: Maintenance Level I Area

Title: Inter-Tank Access Platforms Horizontal

Purpose: Provides cat walk support to the annulus formed between the two propellant tanks. Access will be required thru the outer vehicle structure access door into the area between the two propellant tanks for inspection, removal, installation or checkout of the fluid lines and components located in this area.

Source: New

Location: Maintenance Level I Area

Handling

Title: Protective Cone - Forward

Purpose: Provides Tug, forward structural rigidity for ground operations involving transportation, hoisting, and rotation. Provides overhead protection when Tug is mounted vertically.

Source: New

Location: Maintenance Level I Area, Launch Tower (Contingency Only). Post Manufacturing Checkout, Static Firing and Thermal Vacuum Testing

Title: Aft Skirt Transportation/Storage Pallet - Ground

Purpose: Provides means for storing, transporting and performing maintenance checkout on the Tug aft skirt as a separate assembly.

Source: New

Location: Post Manufacturing Checkout and Maintenance Level I Area

Title: Tug Transportation/Storage Pallet, Vertical, Ground

Purpose: Provides means of storing, transporting and performing maintenance on the Tug in a vertical position. Vertical positioning of the Tug will be required for tank entry operations.

Source: New

Location: Maintenance Level I and Storage Area

Title: Transportation Trailer, Horizontal

Purpose: Provides means of transporting, securing, and performing maintenance operations on the Tug. Provides horizontal Tug rotational capabilities for inspection and maintenance as well as station to station transportation on the ground. Trailer further provides the means of transporting, securing and handling the Tug during air shipment (site-to-site). Trailer will be designed to meet mobility requirements of MIL-M-8090 and air transportability requirements of MIL-A-8421.

Source: New

Location: All sites from post Manufacturing checkout through maintenance Level I and including the safing area and alternate landing sites.

Title: Lifting Sling, Aft Skirt

Purpose: Provides capability to lift Tug aft skirt (ground and flight). Sling trunion and adapter provide rotational capability thru trunion axis, (360°).

Source: New

Location: Post Manufacturing Checkout and Maintenance Level I Areas

Space Division
North American Rockwell

Title: Lifting Sling, Tug-Aft

Purpose: Provides capability to lift Tug aft section in horizontal lifting attitude and thru transition to vertical attitude.

Source: New

Location: Post Manufacturing Checkout, Static Firing, Thermal Vacuum Testing, Maintenance Level I and Alternate Areas as required.

Title: Lifting Sling, Tug-Forward

Purpose: Provides capability to lift Tug forward section in horizontal lifting attitude thru transition to vertical attitude as well as vertical lifting capability.

Source: New

Location: Post manufacturing checkout, static firing, thermal vacuum testing, maintenance level I and alternate areas as required.

Title: Protective Skirt - Aft

Purpose: Provides Tug, aft structural rigidity for ground operations involving transportation, hoisting and rotation. Replaces flight skirt for ground operations and provides integrally mounted GSE systems required during ground checkout operations and static firing.

Source: New

Location: Post Manufacturing Checkout, Static Firing, Thermal Vacuum Testing, Maintenance Level I and Transportation.

Title: Payload Exchanger Adapter

Purpose: Provides capability to support and adapt payload (Tug), to LUT mounted payload exchange equipment. Equipment would be utilized for vertical, contingency loading or off-loading of Tug from the Orbiter on the LUT.

Source: New

Location: Launch Tower (on contingency basis only)

Title: Engine Handling/Removal Unit

Purpose: Provides capability to support and align engine during installation and removal operations. Unit provides means of transporting a single engine from the maintenance level I area to a maintenance level II or III area.

Source: New

Location: Maintenance Level I Area

Checkout

Title: Pneumatic Checkout Console

Purpose: Provides capability to supply, control and monitor ground pneumatics to the Tug vehicle systems during maintenance and checkout operations. Unit provides auxiliary supplies for isolated systems pressurization, purge and checkout operations.

Source: New

Location: Maintenance Area I and Post Manufacturing Checkout

Title: Halogen Charging Unit

Purpose: Provides source of refrigerant gas to recharge the fuel cell heat exchanger refrigerant (Freon).

Source: New

Location: Maintenance Area I, Static Firing Post Manufacturing Checkout

Title: Leak Detector, Halogen

Purpose: Provides portable hand-held leak detection unit for locating and detecting leaks in the fuel cell heat exchanger system.

Source: New

Location: Maintenance Area I, Static Firing and Post Manufacturing Checkout

Title: Pneumatic Leak Detector

Purpose: Provides portable leak detection unit capable of identifying and measuring helium leakage from Tug pneumatic and fluid systems. (Mass spectrometer).

Source: New

Location: Maintenance Area I, Static Firing and Post Manufacturing Checkout

Title: Pneumatic Blanking Plate Set

Purpose: Provides capability to isolate portions of fluid systems or components for isolated system leak checks. Plates or adapters can be installed in lines at component interface or line joints for the purpose of isolating pressure to a portion of a system and/or provides a means of applying pressure to a system or component similarly isolated.

Source: New

Location: Post Manufacturing Checkout, Static Firing and Maintenance Level I.

Title: Pressure C/O Transducer Set

Purpose: Provides higher accuracy ground checkout transducers rated for various pressure levels to supplement flight transducers in conducting diagnostic checkout and fault isolation.

Source: New

Location: Post Manufacturing Checkout, Static Firing and Maintenance Level I Area.

Title: Pneumatic Flow Monitoring Unit

Purpose: Portable unit provides capability to monitor gas flow through systems and components to verify allowable purge flows and component leakage during diagnostic checkout and fault isolation.

Source: New

Location: Post Manufacturing Checkout, Static Firing and Maintenance Level I Area.

10.2 ELECTRICAL SUPPORT EQUIPMENT

INTRODUCTION

This section identifies the electrical support equipment (ESE) required to support the Tug program from hardware development through preflight readiness and launch countdown. The overall objective of the support equipment is to provide that capability required to verify the functional integrity of the flight hardware as it advances from manufacturing through the various phases of the Tug vehicle test program. The support equipment requirements are based upon the subsystem performance criteria and the detailed requirements of the Tug program. The major program function utilized in establishing the types of ESE were flight article manufacturing, acceptance, verification, and post flight turn around.

10.2.1 Program Factors Governing Support Equipment Requirements

The factors dictating support equipment performance are generated by three sources. These sources are (1) Vehicle subsystem operational parameters, (2) The established program test phases and (3) The philosophy adopted for the implementation of the support equipment design.

The operational requirements of the vehicle functions provided by the Tug avionics subsystem are the primary driver in the identification of electrical support equipment requirements. The Tug avionics subsystem require external equipment or conditions to provide the stimulation required to initiate operation during functional testing. This is particularly true of the optical reference equipment provided in the GN&C subsystem when functioning in a non-space environment. The DMS computer requires ground support to provide for the initial loading of operational programs and monitoring during operational verification to insure that the intelligence contained in the DMS computer is correct and also to certify the onboard computer as the central control element of the Tug vehicle systems.

The communication system establishes the requirement for support equipment to provide the means of closing the loop for the airborne telemetry transmitting equipment. That is to provide the means of receiving Tug telemetry data, process it, and establish the validity of the data transmitted. This system also requires ground initiation of uplink data to verify the uplink capability and to stimulate the override functional capability provided by the uplink decoding system.

The power generation function of the Tug is of such a nature that it will not be utilized to provide the basic vehicle power during all phases of testing. This generates the requirement for bas ic power and power control capability in the form of support equipment.

Table 10.2-1 contains a listing of the Tug avionics subsystem with the modules or elements comprising these subsystems and a statement of the type of support the subsystem requires during functional verification.

Table 10.2-1 Tug Avionic Subsystem Test Support

Avionic Subsystem	Test Support
Data Management Computer Data Bus Elements Software	Initial program loading and verification. Perform functional verification of the operating performance of command and control functions. Verify the Data gathering and processing operations. Certify software programs.
Electrical Power Fuel Cell Batteries Inverter Power Distribution & Control Circuitry	Subsystem service and functional performance verification of fuel cells operation and output regulation. Functional verification of secondary power, and control distribution circuitry. Provide external power source.
Communication Antenna - S-band Hybrid Junction Isolation Filter Power Amplifier RF Multiplexer Transponder PM Transmitter, FM Modulator, Bi-Phase Decoder RF Switch Antenna, Rendezvous Power Divider Transponder, Rendezvous	Perform functional tests to demonstrate operating performance and interfaces to verify carrier and subcarrier frequencies, modulation techniques and power levels. Perform loop verification of the uplink and down-link telemetry functions. Provide for processing and evaluation of telemetry data.

Table 10.2-1 (Continued)

Avionic Subsystem	Test Support
Instrumentation	
Pressure Transducers Voltage and Current Monitors Flow Meters Point Sensors Temperature Transducers Signal Conditioning Circuits Position Indicators	Perform functional verification and calibration of instrumentation sensors and signal conditioning to demonstrate operational integrity of system.
Guidance Navigation & Control	
Inertial Reference Units Inertial Measurement Unit Back-up Stabilization Assy Optical Reference Units Horizon Edge Tracker Gimbaled Star Tracker Television Camera Scanning Laser Radar Autocollimator Attitude Control Back-up Stabilization Assy ACPS & MPS Engine Drivers and Servo Amps	Perform functional tests to demonstrate operating performance and interfaces with peripheral equipment. Provide for calibration of guidance and navigation sensors. Demonstrate the ability of the optical equipment to supply valid reference data. Conduct tests to verify the control equipment is operating per design.
Subsystem Sequencing & Control	
Main Engine ACPS Pressurization Propellant Feed, Fill & Drain Propellant Orientation Safing Propellant Management	Perform functional tests to demonstrate operating performance and interfaces with DMS control and monitoring equipment. Demonstrate ability to sequence subsystems as prescribed by mission requirements.

The second major contribution to the performance requirements of the support equipment is the test program established for the Tug vehicle. The Tug avionics equipment will be subjected to the following test phases: (1) Development, (2) Design verification, (3) Acceptance, (4) Flight readiness/turn-around, (5) Prelaunch.

Development testing is that testing conducted to select and prove the feasibility of design concepts. Development testing is concerned with engineering evaluations of hardware, software, and manufacturing processes and techniques for the purpose of acquiring engineering data, identifying sensitive parameters, evaluating the development configuration performance, and providing the necessary confidence that the hardware will mmet the specification requirements and the manufacturing process will produce an acceptable product. Development testing encompasses materials selection and characterization, process evaluation, design feasibility determination, and overall vehicle design and configuration verification, including that for major test article and model tests.

Design verification testing is concerned with engineering evaluation of hardware and software at the subsystem, systems, and integrated system level for the purpose of acquiring engineering data, evaluating overall configuration performance and providing the necessary confidence that the system will meet the specification requirements.

Qualification tests are conducted to verify functional performance and confirm design specification compliance. Qualification tests include functional performance testing to demonstrate the operational suitability of the hardware, static/dynamic testing to verify the ability of the hardware to withstand mission load environments, and thermal vacuum testing to ensure effective performance in a space environment.

Acceptance tests are defined as those tests conducted on deliverable flight and support equipment to demonstrate that the product complies with specifications, is free from defects, and is capable of performing in conformance with stated contractual requirements. Acceptance tests begin with suppliers' test and continue through demonstration at the time of the Space TUG vehicle delivery and acceptance by the customer.

Flight Readiness tests are defined as those verification tests conducted on flight hardware to demonstrate that the vehicle is in an operational state suitable for the performance of mission objectives. Flight readiness tests are conducted prior to each mission. These tests begin prior to the start of flight countdown and continue through to the point of installation of the Tug in the EOS cargo bay.

Prelaunch operations are those subsystem statusing functions performed while the Tug and EOS are in countdown preparatory to launch. The functions are performed utilizing the control panel on-baord the shuttle and in conjunction with the support equipment provided to launch the EOS vehicle.

That specialized laboratory type testing associated with the development of hardware will be conducted by the supplier of the individual components or units. The test equipment utilized in this development will consist of the standard electronic laboratory testing capabilities and some special development devices to facilitate the manufacturing of individual components of the avionics equipment.

Although this equipment is required to support the Tug testing, it is excluded from that equipment classified as support equipment.

Special test facilities and labs are required to provide for the design verification and qualification of hardware to be utilized by the Tug avionics subsystems. The facilities include such as guidance and control development facilities and battleship vehicle. For the most part these encompass existing facilities specialized to perform the desired function and only require the support of a few special developed devices to adapt the facility to the new product. The identification of this equipment requires further definition of the overall Tug program objectives.

The support equipment required to support acceptance preflight and launch operation is an integrated set of specialized equipment designed to certify the functional integrity of the Tug flight system. This equipment comprises the support equipment for the Tug vehicle.

10.2.2 Support Equipment Configuration

The SE will be designed to work through the on-board data management equipment for the stimulation and control of the Tug onboard subsystems. It will utilize the up-link function, telemetry down-link, direct interface with the data bus, and direct access to the DMS computer as the means of communicating with and monitoring the Tug operational performance. Except in cases where the capability does not exist in the on-board equipment to initiate checkout, the ground support function will operate in a passive mode. These exceptions include the verification of the telemetry down-link data and the initiation of vehicle control through the up-link. The support equipment will also be designed to provide that simulation required to support vehicle testing.

The support equipment will consist of a ground computer complex interfaced with the Tug via a direct access connector to the onboard computer, the data bus, and through the communication subsystem utilizing the up-link functions. The support equipment will contain the capability to interface with the communication subsystem via the on-board telemetry capability. This interface will consist of both RF and of a hard line connection down stream of the onboard RF transmitter. Equipment will be provided to provide for the formatting of data and transmitting of this data to the Tug via the up-link function. The performance of this function will also be via RF transmission or via direct access to the on-board equipment. The support equipment will also provide for monitoring of all power voltage and current and any function which may represent a hazardous condition.

Time of day will be provided in the test complex to facilitate the coordination of various data sources and for the recording of timed events during testing.

The ground complex will also provide the means of integrating the operation of the ground support equipment and Tug onboard systems.

That simulation equipment required to stimulate the on-board system will also be provided in the support equipment. The equipment will provide multistar simulation for the on-board star tracker. It will contain earth pattern horizon simulation to verify operation of the on-board horizon sensor. The checkout complex will contain a theodolite for verification of the alignment between the star tracker and the horizon tracker and performance verification of the autocollimator. Equipment will also be provided to implement simulation of the rendezvous and docking functions. In addition to these capabilites, consideration is given to including a method of flight attitude positioning the vehicle to more thoroughly certify the simulated mission activity.

The primary source of support equipment will be the adaptation of existing GSE to the maximum extent possible to perform checkout functions. Between the support equipment designed for the Apollo program and that specified for the EOS program, the capability to checkout the Tug vehicle exists. However, due to site location and mode of operation, it may be practical to design some new support equipment for the Tug program. Any support equipment designed for the Tug vehicle will be subjected to specific design goals. The design goals of the Tug SE are: (1) modularity, (2) ease of adaptability, (3) software flexibility, (4) optimum utilization of automation in order to minimize test personnel and operating time, (5) compact, (6) ease of relocation, (7) minimum facility support and to include provision for remote display or storage, (8) the interfacing and data handling approach adopted for the support equipment will be similar to that utilized by the on-board data management subsystem.

The following contains a brief description of the various elements of the checkout equipment:

Computer Complex

The purpose of the computer complex is to provide the means of controlling the ground operations and monitoring the vehicle subsystem performance during ground checkout. This equipment will provide the man-machine interface required in the performance of automatic checkout and will consist of the computer and that peripheral equipment required to service both the ground and on-board computer. This includes (1) tape decks, (2) card readers (3) line printers, (4) alphanumeric displays and (5) buffering equipment.

The computer complex provides the following functions:

- Provide preprogrammed sequences for subsystem operation.
- Provide integrated control of associated support equipment and vehicle subsystems.

- Monitor GSE and vehicle system responses
- Evaluates system responses
- Provide visual readout and recorded records of test procedures and results.
- Provide self-test capability for GSE.

ELECTRICAL CONTROL DISTRIBUTION SUPPORT EQUIPMENT

The electrical control distribution support equipment provides the means of generating electrical control stimuli for the control of the GSE.

The electrical control distribution support equipment provides the following functions:

- Provides the control, stimuli, response displays and measurement capability required to perform electrical checkout of all systems.
- Provides the interface for checkout of the vehicle system under ground fault isolation.

TELEMETRY SUPPORT EQUIPMENT

The telemetry support equipment performs the checkout functions for the vehicle airborne telemetry system. It also provides the checkout complex with the capability of processing and recording of telemetered data for post test evaluation. The PCM equipment section of the telemetry equipment recovers the pulse code modulated (PCM) data from the vehicle. This equipment processes data to a format acceptable to the ground computer and for display in the complex.

The telemetry support equipment provides the following functions:

- Formatting and routing of PCM data
- Providing data for computer access
- Receiving, monitoring and detecting of the RF carrier frequencies.
- Decoding of telemetry data
- Control of vehicle RF equipment
- Provide for the magnetic recording and playback of the intelligence transmitted via the telemetry system.

UP-LINK SUPPORT EQUIPMENT

This equipment will provide the means of translating computer data into a format suitable for transmission to the vehicle via the up-link system.

This equipment provides the following functions:

- Transmission of computer data to the vehicle via the up-link function.
- Provide the checkout function for the vehicle up-link override capability.
- Process ground computer data to a format suitable for transmission via the up-link function.

REMOTE POWER DISTRIBUTION

The complex will contain the provision for simulating the vehicle basic power. This function will provide from ground sources the power necessary to operate the vehicle subsystem independent of the vehicle power generation capability. It will provide for the distribution and control of these remote power sources. It also provides isolation between the vehicle equipment and any support equipment as required including any protection of vehicle busses utilizing fuses or other means of circuit protection.

TIME CODE GENERATION

The time generator is required to provide time code outputs for real time data recording. The time code also provides the means of time synchronization throughout the GSE complex.

MONITORING

Throughout the checkout complex various monitoring devices are utilized to maintain surveillance of potential hazardous conditions. These monitoring devices will include such equipment as oscillograph recorders to maintain a continuous record of all basic power parameter associated with the vehicle operational systems, and audio-visual alarms to monitor hazardous functions and also to provide facilities control of these functions.

SIMULATION EQUIPMENT

The simulation equipment will provide with the Tug support equipment those capabilities necessary to perform the following functions:

- Multi-star simulation
- Earth pattern horizon simulation
- Rendezvous and docking simulation

- TV camera targeting
- Flight attitude positioning simulation
- Inertial platform alignment fixtures

GROUND SOFTWARE

In addition to ground support hardware, software is also required to mechanize automatic checkout and data reduction. The ground software will consist of the following elements:

- Tug subsystem verification program
- Past test data verification
- Data processing
- Support equipment verification

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11.0 RELIABILITY

11.1 RELIABILITY APPORTIONMENT

A preliminary reliability Apportionment was performed at the subsystem level utilizing a single six-day mission Reliability goal of 0.975. This goal was established as a result of extrapolation from the Apollo and Saturn programs to 1976 technology. Subsystems were weighted by a complexity index which was assigned with due consideration of the number, function and interfaces of components comprising the subsystem. The results of this apportionment effort are presented in Figure 11.1-1.

11.2 SUBSYSTEM FAILURE MODE EFFECT ANALYSIS

A failure mode effect analysis (FMEA) was performed on the functional subsystems considering the loss of subsystem functions and analyzing contributing component malfunctions and the potential effect of these on orbiter crew, payload and payload/Tug mission objectives. The results of the FMEA's are shown in Table B-1 located in the appendix. Criticality of the failure modes was assigned on the basis of the definitions noted in Figure 11.2-1. The analysis, and the associated assignment of criticality considers only single failure points. Single failure points with an orbiter crew hazard and a payload destruction potential present areas for design redundancy to relegate the failure mode to a second order condition. Wherever possible less severe effects such as loss of payload/Tug objectives have been considered for redundant designs commensurate with an analysis of mass fraction impact, reliability, development status and cost effectiveness.

11.3 EFFECTS OF MAINTENANCE ON RELIABILITY

The effects of maintenance on tug apportioned reliability is depicted in Figure 11.3-1. Total cumulative reliability for an unmaintained tug results in a reliability value of 0.603 at the conclusion of 20 missions. Assuming maintenance of the functional subsystems (excluding structures thermal control, wire harnesses and plumbing) yields a reliability figure of 0.959 at the 20th mission.

11.4 PROJECTED RELIABILITY INCREASE WITH 1976 TECHNOLOGY

An effort has been made to project reliability growth for Avionics equipment into the 1976 period based on past growth history. Figure 11.4-1 and 11.4-2 present the decrease achieved in the failure rate of representative equipment, a Minuteman type computer and a typical amplifier. These curves reflect the change from discrete components to transistor and from transistors to microcircuits.

The computer has a 1969 failure rate of $40/10^6$ hrs and it is projected to a 1976 level of approximately $5/10^6$ hrs. This implies a threefold decrease in failure rate. The communications type amplifier indicates a 1968 level of four failures per million hours and is projected to a 1976 level of $0.86/10^6$ hrs, again an approximate threefold decrease. From these two examples a threefold decrease in Avionics failure rates due to component improvement has been projected. This decrease will raise the Tug Avionics reliability from 0.8029 to approximately 0.9294 without recourse to redundancy.

Detailed design effort in the following Tug program phases will optimize reliability and performance by addition of redundancy. Redundant units and/or components will be added at critical design points and a performance reliability trade-off curve will be generated showing the gain in probability of mission success vs. the loss in performance due to weight increase and subsequent decrease in mass fraction.

A preliminary analysis indicates that added redundancy in two areas of the Avionics subsystems will greatly enhance mission success probability. These two areas are as follows:

Option 1 involves the addition of a standby redundant Data Acquisition and Interface Unit that would channel only mandatory and critical vehicle commands and data. This would result in a degraded operational mode for the Tug mission, e.g., a less precise trajectory due to navigation update loss or loss of non-essential vehicle status data. Option 2 would add a standby redundant computer to option one thus improving success probability by eliminating a major single point failure item, the computer. These two options and the resultant gain in mission success probability are depicted in Figure 11.4-3.

Growth in reliability for the mechanical systems, through 1976, is not expected to be as great as in the Avionics area. However, failure rates utilized in preliminary predictions are those derived from S-II components which, while quite similar in design, are considerably larger in physical size which is a significant contributor to higher failure rates. It is also evident that considerable experience has been obtained in the design of components used with LH_2. This design experience when applied to Tug design can obviously be used to improve the operational reliability of the propellant subsystems. The main propulsion engine for the Tug is considered to be a major contributor towards reducing the reliability of the mechanical subsystems.

Reliability follow-on effort will include the development of an overall Reliability Plan which will outline the specific areas, as a function of criticality, where controls will be applied to assure the achievement of the required reliability. Trade studies will also be performed to enable the selection of those areas wherein design redundancy will be utilized to yield highest possible vehicle reliability with least possible impact on the vehicle mass fraction.

A Reliability qualification test plan for components was developed to utilize the integrated test concept. This plan is included in the overall test plan.

Data Control and Management	.9966
Communication and Instrumentation	.9970
Guidance, Navigation and Control	.9970
Rendezvous and Docking	.9973
Main Propulsion	.9976
Auxiliary Propulsion	.9980
Power Generation and Distribution	.9983
Thrust Vector Control	.9983
Pressurization and Insulation Purge	.9987
Propellant Feed	.9987
Propellant Management	.9990
Safing and Vent	.9990
Fuel Cell Thermal Control	.9993
Single 6-day Mission Reliability Goal	.975

Extrapolated from Apollo and Saturn programs to 1976 technology subsystems weighted by component complexity and function.

Figure 11.1-1. Subsystem Reliability Apportionment

CRIT IA	Any single point (first order) failure which will result in injury or loss of the orbiter crew.
CRIT IB	Any single point (first order) failure which will result in physical destruction of payload.
CRIT IIA	Any single point (first order) failure in the tug which will result in loss of the payload mission objectives.
CRIT IIB	Any single point (first order) failure which will result in an unsafe condition for the orbiter crew and vehicle or which will cause loss of tug mission objectives.
CRIT III	All others.

Figure 11.2-1. Tug Criticality Definitions

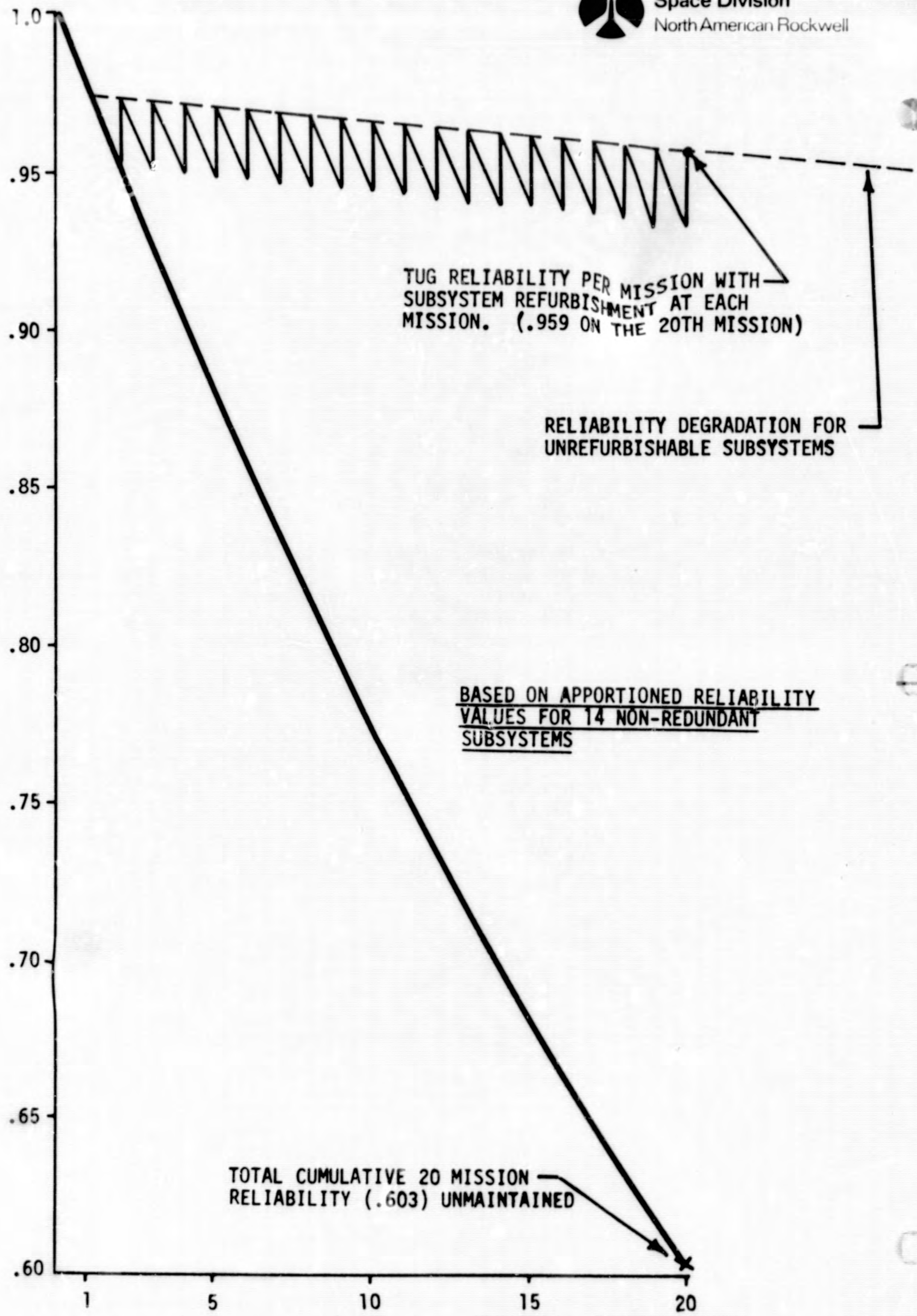

Figure 11.3-1 Effects of Maintenance on Apportioned Tug Reliability

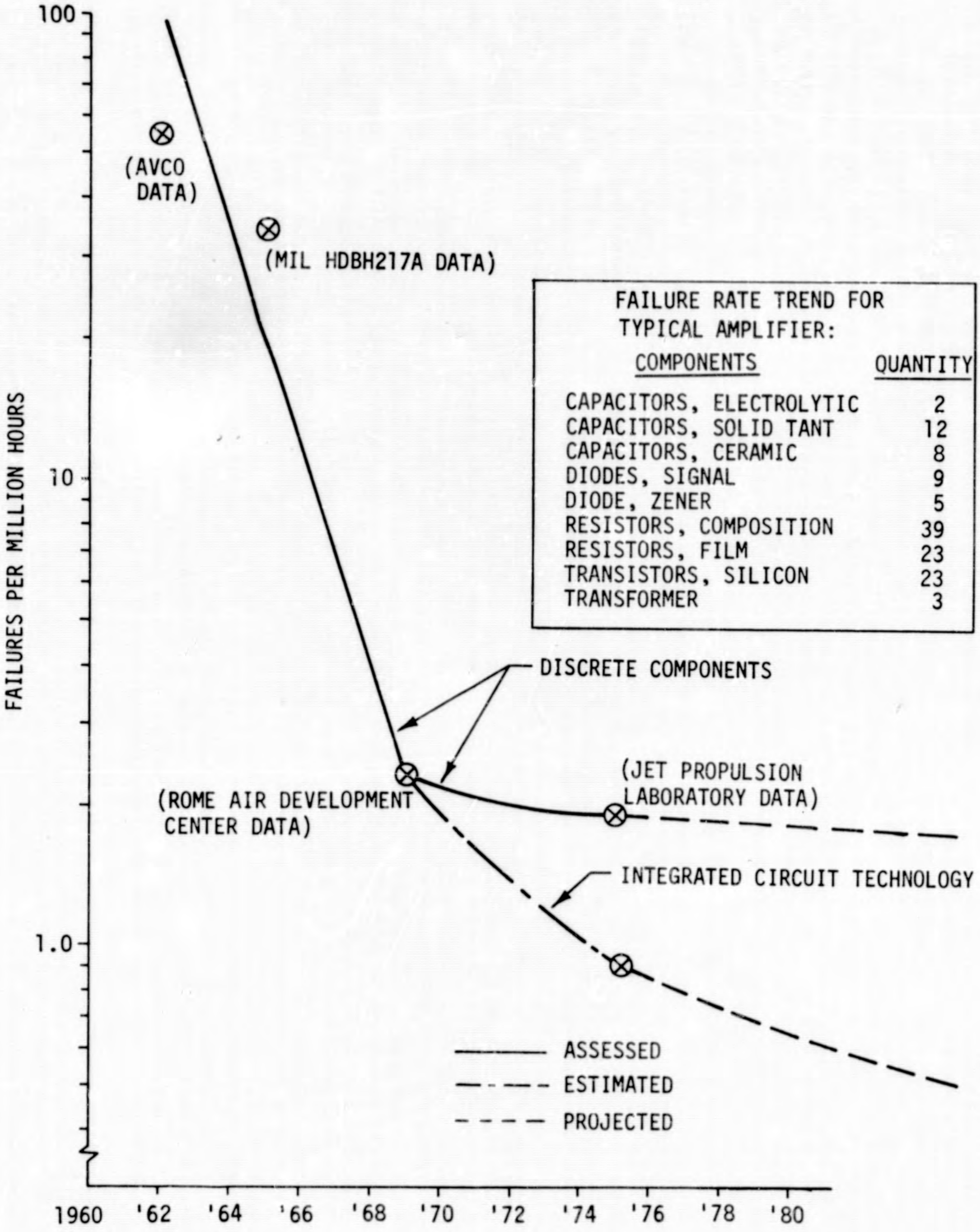

Figure 11.4-1 Failure Rate Trend - Amplifier

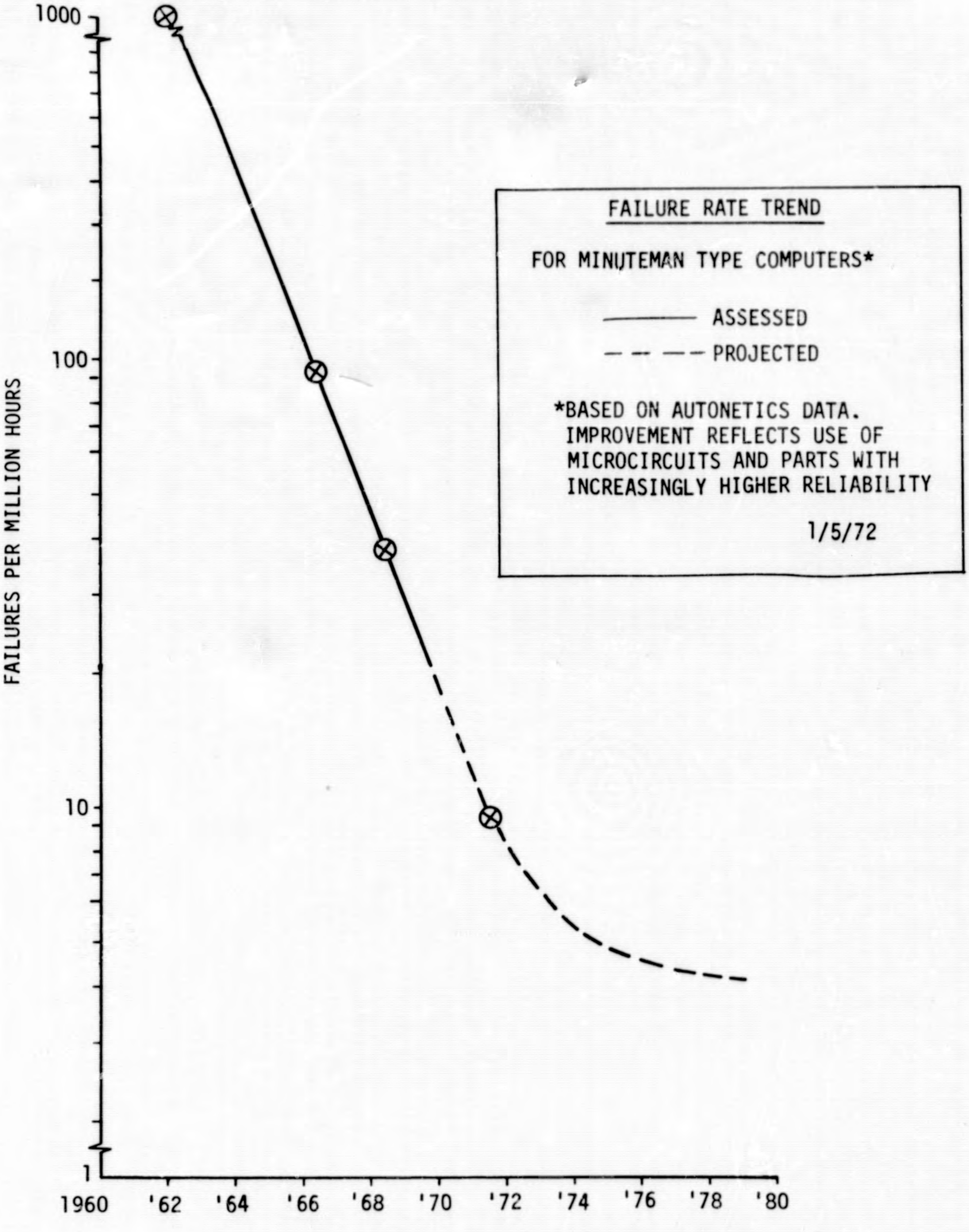

Figure 11.4-2 Failure Rate Trend - Computer

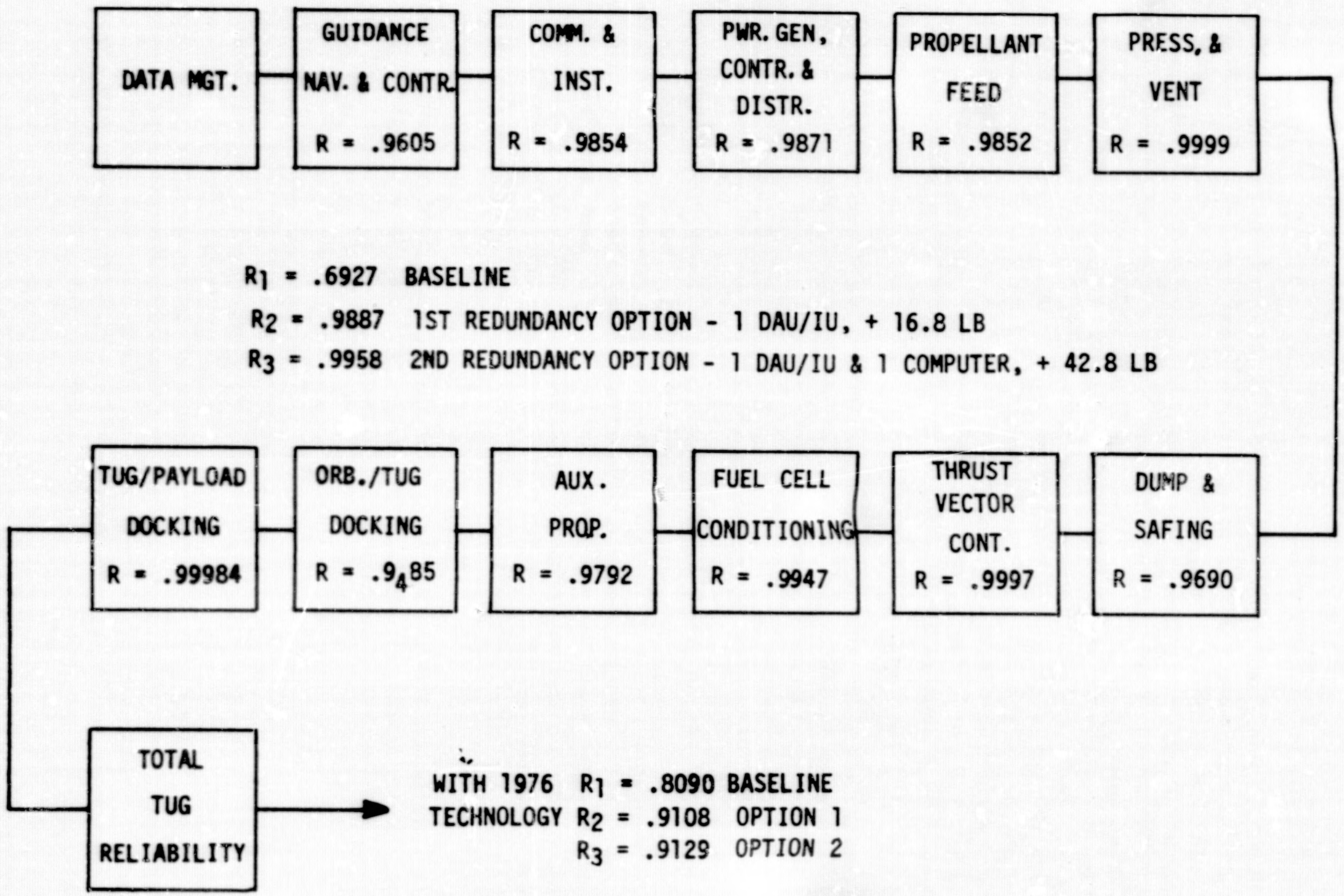

Figure 11.4-3 Mission Success Probability Tug Prediction

11.5 RELIABILITY EVALUATIONS BY SUBSYSTEM

11.5.1 Data Control and Management

The Tug Data Management Subsystem (DMS) is required to integrate, manage, and control the tug vehicle during operation. These functions will include intra-vehicle communications, operational control, uplink control, vehicle sequencing, guidance, navigation, data management, on-board checkout, and statusing. The basic control system will consist of a Central Processing Unit (CPU) and Memory Unit. Information and commands will be transferred through the vehicle by means of Interface Units, Data Acquisition Units, and The Data Bus. There will be a Measurement Processing Unit to provide interface between the vehicle and the Shuttle or Ground up and down link communication. The DMS satisfies the study requirements for fail safe operation through self-check programming and uplink override capability. Use of solid state circuitry, selected redundancy, multiple success path, and alternate mode operation will be evaluated in the design phase to achieve a high mission success probability.

The Failure Mode Effect Analysis contained in Table 11.2-1 (Appendix B) Pages B-1 through B-50 identifies the system functions, components, failure modes and their effects upon the vehicle mission.

11.5.2 Power Generation and Distribution Subsystem

The Power Generation and Distribution Subsystem (PG & DS) consists of a Fuel Cell and its control components, a backup battery, power control switches, and a power transfer switch plus appropriate blocking diodes and load resistance. The PG and DS will provide vehicle power for a six day space mission and 30 minutes of emergency power for vehicle stabilization to meet fail safe requirements. The power control and transfer switches have adequate reliability for for the tug mission; however, the fuel cell and its associated control components will be developed to include internal redundancy for insuring that a high reliability is attainable.

Power Generation and Distribution System component failure modes were considered and are noted in the Failure Mode Effect Analysis contained in Table 11.2-1 (Appendix B), Pages B-17, -18, and -19.

11.5.3 Guidance, Navigation, and Control Subsystem

The Tug Guidance, Navigation and Control Subsystem (GN&C) is required to provide vehicle attitude, position, and velocity information to enable the tug deliver and recover payloads from lower earth to plane changed synchronous orbit and back. The GN&C consists of an Inertial Measurement Unit (rate gyros, accelerometers, and electronics), Star Tracker, Horizon Edge Tracker, Laser Radar, Autocollimator, Television, and Backup Rate and Gyro with associated logic. The Inertial Measurement Unit provides short term state vector and angular rate information. The Star Tracker and Horizon Sensor

provide local earth vertical and state vector update information. The Autocollimator provides inter-sensor alignment for the star tracker and horizon sensor to accommodate vehicle distortions. The Laser Radar and Television provide range, range rate, azimuth, and elevation data for rendezvous and visual contact for docking. The backup stabilization assembly provides 30-minute vehicle stabilization for the fail safe requirement.

System, component, and internal part redendancy, component life development, and alternate mode operation shall be optimized for maximum reliability. GN&C component failure modes are noted in the Failure Mode Effect Analysis contained in Table 11.2-1 (Appendix B), Pages B-9, -10, -11.

11.5.4 Communications and Instrumentation Subsystem

The Tug Communication and Instrumentation Subsystem (C&IS) is required to provide down-link vehicle status data to the shuttle and ground station, to provide transmission of the television channel for man-in-the-loop docking control, and to provide up-link navigation data and docking control signal reception. The C&IS utilizes an FM transmitter, RF Multiplexer, RF Switch, Power Amplifier, Hybrid Junction and two Omni Antennas that provide wide band TV video transmission and status telemetry for a portion of the down-link requirements. In addition, a phase modulator transponder, isolation filter, command decoder, and bi-phase modulator are used for uplink commands, navigation update, down-link status telemetry, and ranging data. The C&IS also provides limited tug vehicle control in a DMS by-pass mode. The instrumentation portion includes parameter sensors and signal conditioners for tug vehicle status information.

The C&IS failure modes are noted in the Failure Mode Effect Analysis contained in Table 11.2-1 (Appendix B), Pages B-1 thru B-23.

11.5.5 Auxiliary Propulsion System

The Auxiliary Propulsion System (APS) provides reaction attitude control for the Tug vehicle during orbital operations and to impart small ΔV changes to the vehicle when required. The APS system consists of two (2) modules containing five (5) thrusters each and two (2) modules containing two thrusters each. The modules are located on the aft skirt assembly.

Gaseous oxygen and hydrogen supplied from propellant accumulators are used for thruster operation, vehicle fuel cell reactants, and main propellant tank prepressurization. The accumulators are replenished by the supply of liquid oxygen and hydrogen from the vehicle main propellant tanks passing through a propellant conditioning system which transforms the low pressure cryogenic liquid propellant into high pressure gases. The propellant conditioning is accomplished by an assembly comprised of a turbopump, heat exchanger, gas generators, and associated controls.

The operational reliability of the system is fail safe (FS) for the liquid propellant feed system and fail operational/fail safe (FO/FS) for the gaseous propellant feed portion of the system. The liquid propellant feed

subsystem is designed to be fail safe plus 30 minutes of APS capability to hold vehicle in a stable position after the failure of a critical component. The gaseous propellant feed subsystem is designed to meet the FO criteria after the failure of a critical component and to FS after the second failure.

The reliability of the APS will also be concerned with ensuring that difficulty in fault detection would not increase the operational hazard level and that high operational margins are utilized for safety critical performance. With the loss of a single APS thruster, the guidance and control system will provide the necessary control of alternate thrusters to compensate for any attitude errors resulting from the inoperative thrusters and maintain attitude hold of the vehicle.

The APS will be electrically disabled during the orbiter boost phase to preclude the possibility of inadvertant operation during this critical mission phase.

The LH_2 and LOX propellant conditioning system will conceivably become a single assembly for each propellant. The FMEA associated with the APS system has evaluated the principal components within the propellant conditioning system; however, during the design implementation phase, a detailed failure mode cause analysis will be made by the assembly manufacturer. This cause analysis will examine potential failure causes and the identification of design characteristics as to why these causes should not occur. The propellant conditioning system will contain protection provisions from a gas generator over-temperature and turbo-pump over-speed conditions.

Pressure regulation of the gaseous propellant from the accumulators to the thrusters is accomplished by utilizing regulators installed in parallel series arrangement providing pressure regulation redundancy for the GOX and GH_2 systems.

The thruster propellant shut-off valves are series redundant with one of the valves being normally open. This no valve is capable of being operated closed by squib actuation if thruster cannot be shut down with normal operation of the electrically controlled shut-off valve.

The Failure Mode Effects Analysis contained in Table 11.2-1 (Appendix B), Pages B-19, -20, -21 and B-24 through -32 identifies the auxiliary propulsion system functions which could contribute to the degradation of system performance. The FMEA also identifies system components with their appropriate failure modes, their effect upon the system, stage mission performance, and establishes the component criticality category.

11.5.6 Propellant Tank (LH_2 and LOX) Pressurization System

The main propellant tank (LH_2 and LOX) pressurization system consists of bleeding GH_2 and GOX from the main operating engine and routing the gas through a pressure regulator assembly to maintain proper tank ullage pressure control. To ensure engine start, prepressurization of the propellant tanks is accomplished by utilizing pressurized GH_2 and GOX supplied from the APS accumulators.

The operational reliability of the pressurization system is based on the overall vehicles fail safe design criteria. The prepressurization system utilizes two (2) solenoid control valves mounted in series in each of the GH_2 and GOX lines from the APS accumulators. The control valves are normally closed valves and are used in series to prevent loss of the APS accumulator pressure resulting from a single failure of a control valve to completely close. The above design concept retains the fail safe criteria with the 30 minute vehicle attitude hold capability within the APS accumulators.

The Failure Mode Effects Analysis (FMEA) contained in Table 11.2-1 (Appendix B), Pages B-6, -7 and -33 identifies the system functions, components, failure modes, and the criticality category for each of the identified components.

11.5.7 Main Propellant Tank Safing and Venting System

The Vent Safing System maintains necessary ullage pressure in the main propellant tanks for main engine operation and provides for draining and purging the main propellant tanks following completion of the mission for return to earth by the orbiter. The safing portion of the system noted herein is also associated with the He supply and pressure control system used for propellant tank purge and vehicle compartment/tank insulation purge.

The failure criteria applied to this subsystem requires that following any single component failure the resulting effect shall be fail safe. The subsystem as presently defined meets this criteria satisfactorily.

For the purpose of the Reliability analyses where hardware failures results in loss of capability, operational considerations were applied for alternate means of performing the function. This is especially true in utilizing the main propellant tanks fill and drain subsystem to back up normal tank safing operation. The tank safing operation is accomplished in less than one hour. The fill and drain subsystem can be utilized for venting of the tanks to vacuum where time is not critical.

For normal venting and safing operations redundancy was applied to the degree necessary to preclude a single failure from seriously impacting manual operations. In the tank vent loop, the selector valves, vent valves and the associated solenoid valves to the non-propulsive vent are parallel redundant. Parallel redundancy was also applied to the solenoid valves in the fill and drain loop although single failures exist in the pressurization loop which could prevent pressurizing the tank for re-entry. Collapse of the tank during earth return would result.

System component failure modes associated with the various system operational phases were considered and are noted in the Failure Mode Effect Analysis, Table 11.2-1 (Appendix B), Pages B-8, -9, and B-34 through B-40.

11.5.8 Propellant Feed System

The propellant feed system consists of insulated LH_2 and LOX feed lines routed from the stage propellant tanks to the main engine. A prevalve is installed at the interface of the feed line and engine inlet duct in both LH_2 and LOX feed systems. A thermal isolation segment is located between the prevalve and the engine inlet which prevents propellant boiloff during engine shutdown with the prevalve closed. Temperature conditioning of the feedlines is accomplished with a LH_2 thermodynamic vent system. This system circulates LH_2 from the LH_2 APS acquisition tank, around the LH_2 and LOX feed lines and the APS liquid propellant distribution system.

The principal failure mode for the propellant feed system is associated with the prevalve failing to open which would prevent main engine operation, resulting in mission loss. However the fail safe criteria for the vehicle would be met. Failure of the thermodynamic vent to maintain the required temperature conditioning of the main and APS feed lines is not considered to be critical. Some propellant loss would occur through initial boiloff causing early depletion of propellant and possible loss of completing the mission. The principal component failure modes and mission effects associated with this system are identified in the Failure Mode Effect Analysis contained in Table 11.2-1 (Appendix B), Page B-41.

11.5.9 Fuel Cell Thermal Control System

The Fuel Cell Thermal Control System consists basically of four radiators and a freon pump with a water storage/supply system supplying supplemental cooling of the fuel cell during high load conditions. Fail safe establishment of the operational reliability design precautions have been considered for meteroid protection of the radiator panels by an increase in skin thickness. No maintenance is anticipated for the radiator panels during the 20 mission life of the vehicle.

Table 11.2-1 (Appendix B), Pages B-42, -43 contains the Failure Mode Effect Analysis with identifies single point failures associated with the subsystem.

11.5.10 Main Propulsion System (MPS)

The main propulsion system consists of one (1) liquid hydrogen/oxygen engine. The basic engine is equipped with a low speed inducer, turbopump, electronics package, preburner, and necessary valves and related hardware. Any failure that causes loss of engine operation results in a loss of mission. Depending on the magnitude and the time period, an engine explosion could be hazardous to the orbiter and crew. An engine explosion occurring with the payload attached could result in physical destruction of the payload.

A detailed failure mode cause analysis (FMCA) on the engine components will be performed by the engine manufacturer detailing the various engine system components which could contribute to loss of engine failures. The principle single failure modes of the main engine are listed in Table 11.2-1 (Appendix B), Page B-44.

11.5.11 Thrust Vector Control (TVC) System

The Thrust Vector Control System is a conventional electric-powered hydraulic system. The system components are an integral part of the actuator assembly with external plumbing connecting the two (2) actuator assemblies. One actuator assembly contains the inverter, motor and pump, servo valve, and an actuator piston. The other actuator contains the accumulator reservoir, servo valve and piston. Integral component packaging enhances the reliability of the system by eliminating related external plumbing and electrical control cables.

Loss of hydraulic power will result in lock-up of TVC in the last commanded position. However, a system failure resulting in main engine shut down would cause unsuccessful completion of the mission but would meet fail safe criteria for the vehicle. The principal failure modes are listed in Table 11.2-1 (Appendix B), Pages B-23, -24 and B-45, -46.

11.5.12 Insulation Purge System

The insulation purge system provides ground purge of the propellant tank insulation prior to propellant loading and also maintains Tug structure compartment at a positive pressure differential of 0.1 psi during re-entry. System incorporates two (2) vent valves and two (2) pressure switches to control the positive pressure differential. Redundancy in valves and switches meets the tug fail safe criteria. The solenoid valves that control the purge gas to the tank insulation have no flight function and are not considered critical to achieving a high mission success probability.

Components and their related failure modes associated with this system are noted in the Failure Mode Effects Analysis contained in Table 11.2-1 (Appendix B), Pages B-47 and -48.

11.5.13 Docking Systems

The docking system on the tug vehicle is unique in that the forward and aft end both require a docking device. The aft end of the tug is equipped with a docking mechanism that interfaces with the shuttle orbiter while the forward end of the tug has a docking device to deploy and retrieve the payload. The two docking devices are two completely independent systems. The docking system concept is considered representative in weight for a typical docking system; however, the design concept shown will require further design development for improving the operational reliability of the system.

Development of a standard docking system to be compatible with the various payloads and possibly with the orbiter adaptor would enable achievement of a minimum weight design with high operational reliability for all payload/tug combinations.

In pursuit of a standard light weight and reliable docking system, the design concepts will evaluate various type of docking systems. These systems will consider latching devices that contain a single operating mechanism with

appropriate back-up, shear type devices for release of latch mechanisms that have failed to operate, and the possibility of squib operated mechanisms as a backup actuation system.

11.5.14 Orbiter/Tug Docking System

The orbiter/tug docking latching mechanism is at the interface of the tug aft bulkhead and Station 126 of the orbiter interstage adapter. The orbiter interstage adapter acts as the support mechanism for the tug while it is in the orbiter cargo bay, and also provides the means of deployment of the tug from the cargo bay. The mechanism for release of the tug is oriented with the tug interstage adapter. The mechanical actuators and related hardware that extends the interstage adapter, with the tug attached, from the cargo bay is considered an orbiter responsibility and thus not analyzed in this study.

The docking mechanism as presently depicted has a series of latches which must all operate for a successful deployment. For the orbiter/tug separation, four actuators and sixteen latches must operate plus eight individual electric actuated latches. Thus a first order failure would result in mission loss. A tug returning to the orbiter for docking requires two of the three probe and drogue units to operate plus the relatching of the eight electric-actuated latches. Failure of two adjacent latches to operate would possibly result in some tug structural damage during re-entry. Upgrading the reliability by providing redundancy and or other electro/mechanical attachments will be accomplished during subsequent design study efforts. System components and related failure modes are noted in the failure mode effects analysis contained in Table 11.2-1 (Appendix B), Page B-4 .

11.5.15 Tug/Payload Docking System

The docking mechanism at the forward end of the tug which interfaces with the payload also has first order failures which would result in mission loss. Any one latch of twenty-four failing to unlatch would result in failure to place the payload in orbit. Failure of two (2) or more latches to relatch during retrieval of the payload could possibly cause vehicle structural damage during re-entry. System components and related failure modes are noted in the failure mode effects analysis contained in Table 11.2-1 (Appendix B), Page B-50.

12.0 SYSTEM SAFETY

12.1 INTRODUCTION AND SUMMARY

Introduction

In accordance with NASA SPD-1A, "System Safety Requirements for Manned Space Flight," a Preliminary System Safety Analysis was conducted for the Tug point design study. This analysis disclosed two areas that were examined in detail. The first concerned hydrogen leakage in the shuttle cargo bay, and the second addressed the operational requirements of the Tug during sub-robital abort.

Other conclusions and recommendations that arose during the System Safety Analysis are included in section 12.4.

Summary

The Tug safety effort concentration was in three main areas with a strong effort in the final collection of all safety data. The main areas were the hazards analysis, sub-orbital abort, and hydrogen leakage in the cargo bay.

The hazards analysis is the identification and evaluation of potential hazards in the Tug systems and Tug operations. The analysis was conducted with the aid of systems and operations flow diagrams and schematic drawings from the design groups. Systems, operations and hazards analysis from past, similar studies were also used for this analysis. The hazards analysis was reviewed by the various design groups and their comments were incorporated into the final revision.

The Tug sub-orbital abort capability study was a result of a NASA request. Since the Tug is completely dependent on the shuttle orbiter during launch, the orbiter abort capabilities had to be investigated. The results were derived from data obtained through the NR/SD Space Shuttle, Phase B Study.

The hydrogen leakage problem required a concentrated effort. From this investigation system, Safety was able to establish recommendations concerning the leakage of propellants in the shuttle cargo bay.

From the above tasks, final guidelines and requirements were generated for the final report. These requirements are to guide further study and design.

12.2 SYSTEM SAFETY ANALYSIS

In accordance with 1) NASA SPD-1A, "System Safety Requirements for Manned Space Flight," 2) NR Pub 543-9-37, "System Safety Design Standards" and 3) study ground rules, the initial safety guidelines and requirements were developed for the Tug criteria and constraints document, and a Preliminary Hazard Analysis was conducted for the Tug point design study.

A representative functional flow block diagram (Figure 12.2-1) of a typical Tug mission was developed. The block diagram grouped those operations that would be performed into 8 operational segments from pre-launch through orbital operations to re-entry, landing and safing. Using experience gained from past programs and knowledge and data from previous studies, each functional block was examined for hazard potential at the system level. Supporting data from subsystems was used as necessary to establish the system performance. The hazard was identified, its effect was stated and the means established for its reduction or elimination.

This Gross or Preliminary Hazard Analysis disclosed several areas that warrant further study. Two of these areas, 1) suborbital abort and 2) hydrogen leakage in the cargo bay, were studied in greater detail.

Table 12.2-1 details the Hazard Analysis and presents the most significant hazards identified and the means recommended for their reduction or elimination.

As a result of this Hazard Analysis the initial System Safety Guidelines and Requirements were modified for inclusion in the final report to guide future development of the Tug.

12.2.2 Sub-Orbital Abort

The limitations and requirements imposed on the Tug during a sub-orbital abort are determined by the configuration of both the shuttle booster and orbiter. The mission time-lines between lift-off and placing an orbiter and its payload (Tug and its payload) into orbit will vary with each different configuration of these two vehicles. These combinations have been studied by NR/SD during the Phase B shuttle studies.

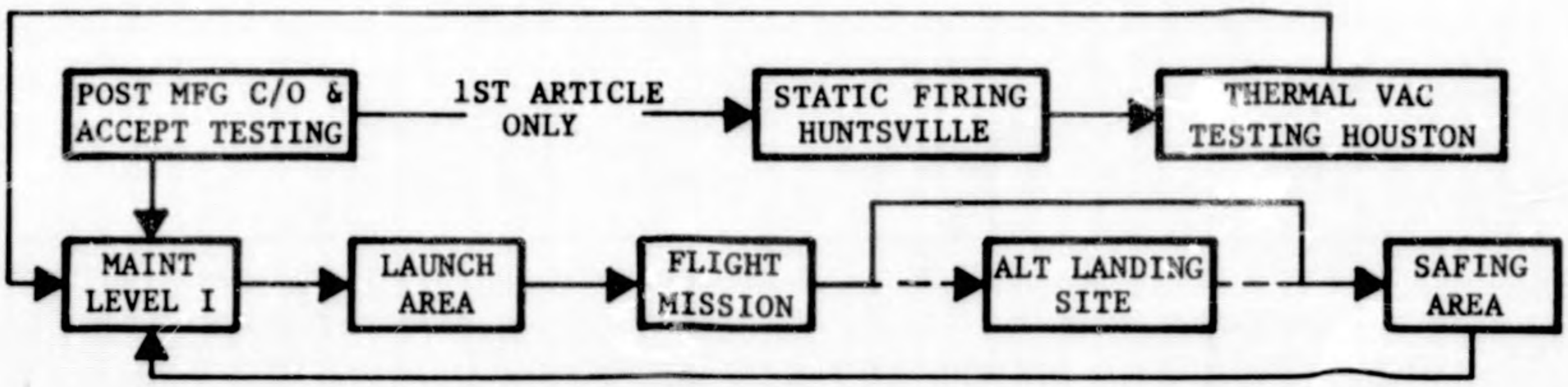

Figure 12.2-1 Tug Mission Functional Flow

Table 12.2-1 Hazard Analysis (Sheet 1 of 4)

Function	System	Hazard	Effect	Preventive Action
I. PRE-LAUNCH/ LAUNCH/ASCENT				
1. Loading Tug into orbiter	1a. Crane	1a. Crane Failure Operator error	1a. Damaged Tug/orbiter " " "	1a. Proofload crane Procedural controls Operator training
2. Propellant Loading	2a. Propellant Connects	2a. Premature Separation	2a. Mixing Propellants Probably fire/ explosion	2a. Redundant controls Interlock to close fill valve Self-closing dis- connect Physical separation of propellant con- nectors Inert cargo bay
	2b. Propellant Connects	2b. Leakage	2b. Fire/explosion	2b. Leak Checks - Cargo bay purge
	2c. Propellant Tanks	2c. Over-pressurization	2c. " "	2c. Redundant vent system
	2d. " "	2d. Excess propellants	2d. Two-phase flow from tank vents	2d. Facility vents designed to handle two-phase flow
	2e. Propellant Loading	2e. Excess leak in bay	2e. Fire/explosion	2e. Inert cargo bay
3. Sub-orbital abort	3a. Structural	3a. High loads on landing orbiter	3a. Structural failure	3a. Design for landing with 8060 lb. pay- load. Provide LOX dump capab. to achieve 40K down limit.
II. TUG DEPLOYMENT	1a. Elect. umbilical	1a. Short circuit due to contamination/ moisture/corrosion	1a. System Degradation Component failure	1a. Overload protection Proper material selection
	1b. Elect. umbilical	1b. Arcing	1b. Elect. Failure	1b. Remove power from umbilical connect- ions prior to separation.
	1c. Elect. umbilical	1c. Does not disconnect	1c. Damaged system	1c. Status devices Multiple attempts

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Table 12.2-1 Hazard Analysis (Sheet 2 of 4)

Function	System	Hazard	Effect	Preventive Action
II. *TUG DEPLOYMENT* (Cont'd.)	2a. Pressurization	2a. Over-pressurization of system or components	2a. Component and/or system rupture	2a. Relief valves
	2b. Pressurization	2b. Inadequate pressure relief system or vent	2b. Explosion	2b. System flow checks
	2c. Pressurization	2c. Burst Disk Blows out	2c. Vents vapors through open area	2c. Route burst disk vent to safe area
3. Propulsion	3a. Fuel cell	3a. Failure to activate fuel cell	3a. Loss of power source (Abort)	3a. Redundant controls
	3b. Propulstion	3b. Premature operation	3b. Loss of control (Abort)	3b. Redundant controls
4. Deployment	4a. Deployment Mechanism	4a. Rotator stuck in open position	4a. Further use of mechanism lost	4a. Redundant controls
	4b. Deployment Mechanism	4b. Multiple usage failure	4b. Possible collapse of structure	4b. Life test
	4c. Deployment Mechanism	4c. Locking latches fail to release	4c. Deployment not possible	4c. Provide alternate release methods
III. ORBITAL INJECTION				
1. GN&C	1a. Computer	1a. Failure	1a. Loss of status knowledge & decision making	1a. Redundant capability
	1b. Inertial Meas. Unit	1b. Malfunction	1b. Navigation impaired	1b. Redundant capability
	1c. Horizon sensor	1c. Malfunction	1c. Meas. information loss	1c. Redundant capability
	1d. Star tracker	1d. Malfunction	1d. Navigation impaired	1d. Redundant capability
	1e. Communication	1e. Damage to Ant.	1e. Communication loss	1e. Redundant capability
	1f. Rate gyros, accelerometers	1f. Navigation error due to failure	1f. Non-recoverable orbit	1f. Redundant capability

Table 12.2-1 Hazard Analysis (Sheet 3 of 4)

Function	System	Hazard	Effect	Preventive Action
2. Propulsion	2a. Engines	2a. Ignition failure	2a. Mission loss	2a. Redundant capability
	2b. Engines	2b. Uneven combustion	2b. Structural failure	2b. Malfunction cutoff
	2c. Engines	2c. Valve failure	2c. Mission loss	2c. Redundant capability
	2d. Engines	2d. MRC valve failure	2d. No return capability	2d. Valve position sensors
	2e. Engines	2e. Turbo machinery structure failure	2e. Tug loss	2e. Control unit sensors
	2f. Engines	2f. Pump cavitation	2f. Mission loss	2f. Assure adequate NPSH
	2g. Engines	2g. Insufficient ullage pressure	2g. Mission abort	2g. Redundant pressurizing capability
	2h. Engines	2h. Bell nozzle burn out - LOX rich concentration	2h. Mission abort	2g. Safety devices in control unit - malfunction cutoff
	2i. Engines	2i. Boost pump failure	2i. Mission abort	2i. Prior inspection
	2j. Engines	2j. TVC failure	2j. Mission abort	2j. Redundant capability
	2k. RCS	2k. Failure to stabilize Tug	2k. Tug loss	2k. Alternate control
3. Flight	3a. Tank structure	3a. Meteoroid bombardment	3a. Damaged tanks	3a. Meteoroid shield
IV. PAYLOAD SEPARATION FROM TUG	1a. Separation Device	1a. Premature activation of device	1a. Premature separation of payload	1a. Use 2 separate commands for this phase
	1b. Release latches	1b. Latches fail to release	1b. Mission abort	1b. Status devices alternate capability
V. RENDEZVOUS WITH PAYLOAD OR ORBITER	1a. GN&C	- - - See IV.1. GN&C, items a-3 inclusive - - - - - - - - - - - - -		
	1b. Laser radar	1b. Malfunction	1b. Loss of ability to locate Tug	1b. Redundant capability
	1c. T.V.	1c. Malfunction	1c. Collision with orbiter	1c. Procedural controls
	1d. Payload	1d. Failure to dock with Tug	1d. Mission loss	1d. Operator training procedural control

Table 12.2-1 Hazard Analysis (Sheet 4 of 4)

Function	System	Hazard	Effect	Preventive
V. (Cont'd)	1e. Payload	1e. Failure to latch	1e. Mission loss	1e. Redundant latch mech.
	1f. Tug	1f. Inability to stabilize	1f. Rendezvous abort	1f. RCS control
VI. TUG PLACEMENT	1a. Dock Mech.	1a. Mech. uncontrollable	1a. Impact tug into bay	1a. Procedural control
	1b. Horiz. sensor	1b. Failure to stow away	1b. Inability to stow Tug	1b. Redundant capability
	1c. Umbilical panel	1c. Failure to connect to interface w/orbiter	1c. Inability to safe Tug	1c. Back-up controls
	1d. Propellants Systems	1d. Unable to dump propellants	1d. Tug unsafe to de-orbit	.d. Refrain from de-orbit until line intercon-nect fixture is engaged
VII. RE-ENTRY	1a. Tanks	1a. Below min. pressure	1a. Implosion	1a. Perform pressure check
	1b. Hydrogen Sys.	1b. Leakage	1b. Explosion potential	1b. Inert bay
	1c. Insulation	1c. Leak of O_2 & H_2 in purge bag	1c. Explosion potential	1c. Leak checks
	1d. Tug secure fixture	1d. Failure to lock	1d. Damage to Tug & bay	1d. Warning device
VIII. GROUND SAFING	1a. Pressurization	1a. Explosive potential	1a. Fire or explosion	1a. Purge system of residual hydrogen
	1b. Hydraulic	1b. Stored energy	1b. Rapid release of pressure	1b. Depressurize systems

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As a result of these studies the shuttle abort options can be divided into four regimes. The first regime occurs from time 0 to 20-30 seconds after lift-off when no abort capability exists for the shuttle. The necessity for abort arises because of a failure of the booster. The orbiter is unable to fly away from the booster because it has not received sufficient velocity from the booster to gain independent flight. Thus, no abort capability exists for the shuttle during the first 20-30 seconds. This time span is variable and will be fixed when the configuration of the booster and orbiter are defined.

The second and third shuttle abort regime occurs between the first abort regime and booster/orbiter staging and again is due to a booster failure. Booster/orbiter staging occurs at different times for different configurations and orbital inclinations but for Tug purposes was chosen as 140 seconds after lift-off. An abort during this period would permit the orbiter to fly away from the booster and land at the launch site for the second abort regime and at a down-range site for the third regime. These two regimes can be classed as one as far as the Tug is concerned since the flight time of the orbiter after separation until landing at either site is the same, 200-300 seconds to burn propellants and drop tanks. This time is again variable because of mission and configuration variances.

The fourth abort regime occurs after booster/orbiter staging and implies an orbiter failure to achieve the desired orbit. Under these conditions the orbiter has once-around abort capability with a flight time of up to 6000 seconds.

Structural limitations have been placed on the orbiter to enable it to land with a 40,000 pound payload with an optimum factor of safety and heavier payloads with a decreased factor of safety. This places the requirement to lighten the gross weight of the Tug to permit it to land during the emergency conditions attending an abort without compromising the safety of the orbiter and crew. The most feasible method of reducing the weight of the Tug would be to dump propellants. Because of the near-vertical attitude of the orbiter and the possibility of propellants lingering near the orbiter hydrogen should not be dumped during this period. In addition the weight of the hydrogen aboard the Tug could not lighten the gross weight the maximum of 25,000 pounds that might be required.

Operational restrictions will limit the time that it will be possible to dump LOX to lighten the Tug. It is not expected that LOX dump could be initiated immediately after separation from the booster. Neither is it expected that LOX will be dumped during the final approach for landing.

If the shuttle abort regimes are divided into three abort modes the first mode will not have the capability of aborting the Tug. During the second mode up to 25,000 pounds of LOX must be dumped in less than 200-300 seconds to optimize the safety of the landing. Even after LOX dump the Tug structure would be damaged if the maximum Tug payload were

aboard the Tug. Structural damage of the Tug could lead to structural damage of the orbiter.

It is only during the third or once-around mode that sufficient time is available to completely dump propellants and safe the Tug in the cargo bay of the orbiter. The same Tug structural limitations still are present when an orbiter landing is made with the maximum payload on the Tug.

These results are summarized in Figure 12.2-2, the Abort Envelope.

12.2.3 A Discussion of Hydrogen Leakage Into Shuttle Cargo Bay

Objective; To develop a technique of establishing the risk potential of Hydrogen leaking into the Shuttle cargo bay from a propellant tank carried as cargo

Technical Discussion: Pre-Launch Case

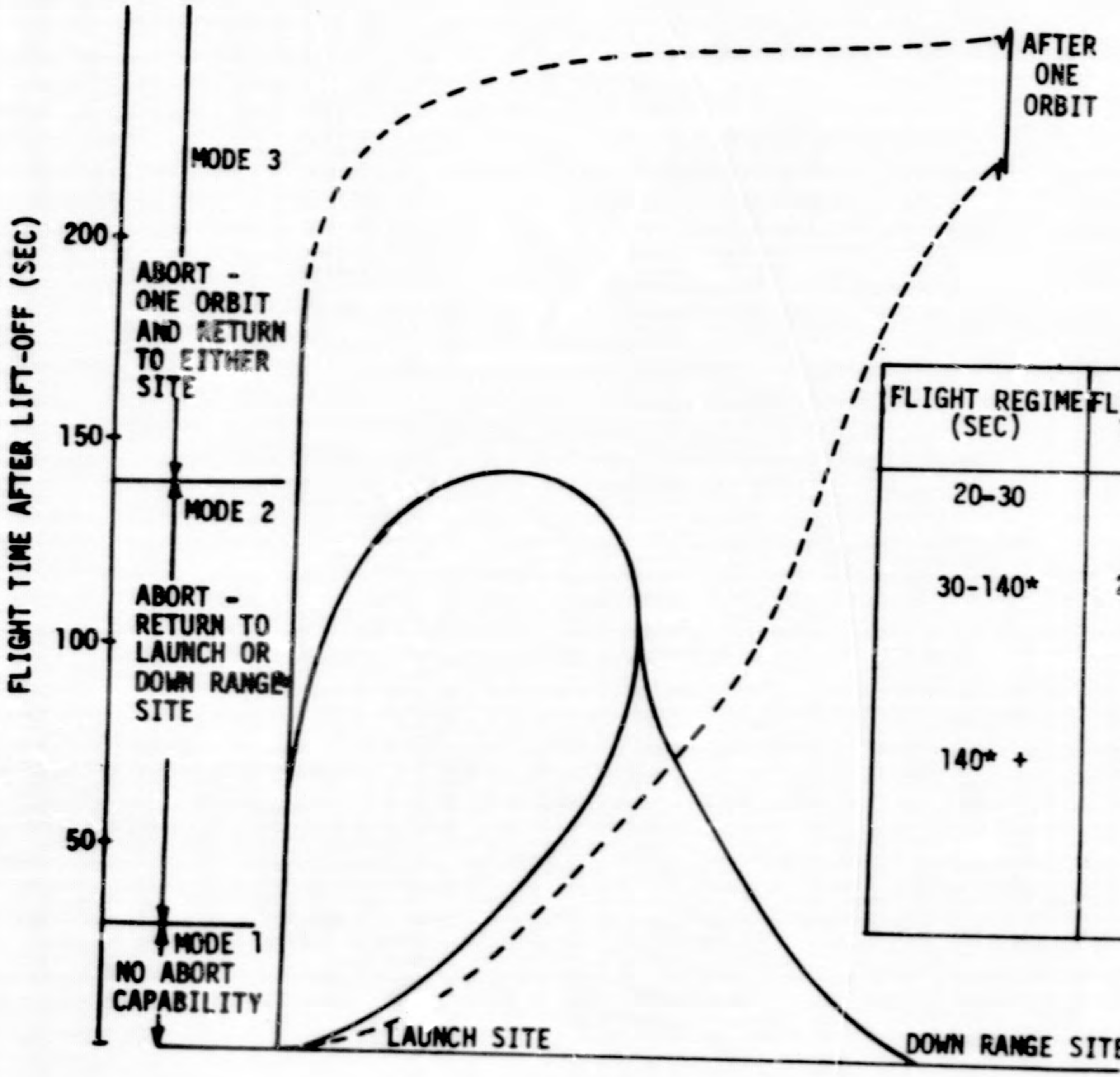
Prior to loading the propellant tanks of a shuttle cargo during the pre-launch phase it can be assumed that ground checkout will include leak checks and be more thorough than that which can be achieved in orbit. Further, the nitrogen purge of the cargo bay will effectively inert this cavity to reduce the risk of a hydrogen-oxygen reaction to negligible proportions. For these reasons the pre-launch case of hydrogen leakage will not be governing.

Technical Discussion: Re-Entry Case

A. Groundrules and Assumptions

If a typical example of a shuttle cargo such as the Tug is chosen for analysis the following groundrules and assumptions can be established:

1. 4% by volume of Hydrogen is the minimum percentage necessary to support a reaction. (Ref. 12-1)
2. Hydrogen-oxygen reactions can occur at pressures equal to or greater than 2 mm Hg. (Ref. 12-1)
3. No liquid Hydrogen remains in the Tug Hydrogen Tank. (Ref. 12-2)
4. Atmospheric oxygen is available for reaction.
5. Assume that every source that can leak hydrogen is leaking at its allowable rate, these leaks are distributed through the entire length of the Tug, and the leaking gas mixes intimately in the volume available to it through molecular diffusion.
6. Any hydrogen-oxygen reaction is undesirable and the energy release is dependent on the mass of reacting hydrogen.



* APPROX STAGING TIME

MODE 1 - BOOSTER FAILURE NO ABORT CAPABILITY

MODE 2 - BOOSTER FAILURE - 200-300 SEC ORBITER FLIGHT TIME-DUMP TUG LOX

MODE 3 - ORBITER FAILURE - ONE ONCE AROUND ORBIT CAP. - DUMP & INERT TUG TANKS

FLIGHT REGIME (SEC)	FLIGHT TIME TO LAND (SEC)	TUG PROPELLANT DUMP ABILITY	SAFETY CONSIDERATIONS
20-30		• NO ABORT CAPABILITY	
30-140*	200-300	• LOX CAN BE DUMPED	• DUMP LOX
		• LH_2 WILL BE KEPT ON BOARD	• SAFETY HAZARD WITH LH_2 DUMP IN ATMOSPHERE
140* +	6000	• BOTH LOX AND LH_2 CAN BE DUMPED (TIME REQ'D TO DUMP BOTH 27 MINS)	• SAFEST ABORT POSSIBLE

Figure 12.2-2 Abort Envelope

B. Determination of Leakage Rate

A typical tug schematic was examined and a total of 233 possible leak points were counted that could be exposed to hydrogen under pressure. These leak points include two 20" diameter manhole covers. Many of these leak points will be welded joints and flared fittings for which the allowable leakage is 5.3×10^{-4} scc/sec. (Ref. 12-3). If the worst case is assumed and it is assumed that all the leak points are flanged fittings 2-1/2" diameter the allowable leakage is 0.01 scc/sec/inch of seal (scc-standard cubic centimeters). (Ref. 12-3). At this leakage rate from 233 flanged fittings a volume of 6.07 scf/day or approximately 1/4 scfh (standard cubic feet per hour) will leak from the Tug.

C. Determination of reaction volume of leaking hydrogen

If the Tug is emplaced in the cargo bay and the doors are closed in preparation for re-entry the volume available for reaction is the net volume of the cargo bay minus the cargo volume. The cargo bay volume is 19,400 cubic feet (Ref. 12-4) and the Tug volume can be calculated as approximately 2700 cubic feet and the volume available for reaction, the free volume. is 16,700 cubic feet. Of course, larger cargoes or payloads on the Tug would reduce the free volume, but not enough to alter the conclusions.

D. Determination of Hazardous Concentrations

During re-entry preparations utilizing the current sequence of events the Tug will be emplaced in the cargo bay, the residual liquid hydrogen will be dumped and the tank vented to vacuum and re-pressurized to 17 psia (one or more times). At this time the shuttle will begin its re-entry flight. The Tug will continue to leak hydrogen gas from the possible leak points and this hydrogen will accumulate in the free volume when external pressure is greater than cargo bay pressure.

Using internal cargo bay pressure from Figure 12.2-3 (Ref. 12-5), the leaking hydrogen that is expressed as standard cubic feet will expand to fill a larger volume. Since computer programs have not been developed to permit predictions of how much of the leaking hydrogen would diffuse out of the cargo bay, the worst case was assumed. If all the leaking hydrogen remained in the cargo bay from time zero (closing cargo bay doors) until cargo bay pressure becomes great enough to permit reactions, the leaking hydrogen will expand inversely as the cargo bay pressure. Several points on the curve in Figure 1 were chosen to investigate the influence of cargo bay pressure and it was found that a constant hydrogen leakage would expand to maximum concentration at 2 mm Hg., where reactions are first possible. As a consequence the single most hazardous time in the re-entry flight profile is that time at which cargo bay pressure reaches 2 mm Hg., and less hazardous before and after that time - as far as hydrogen leakage is concerned.

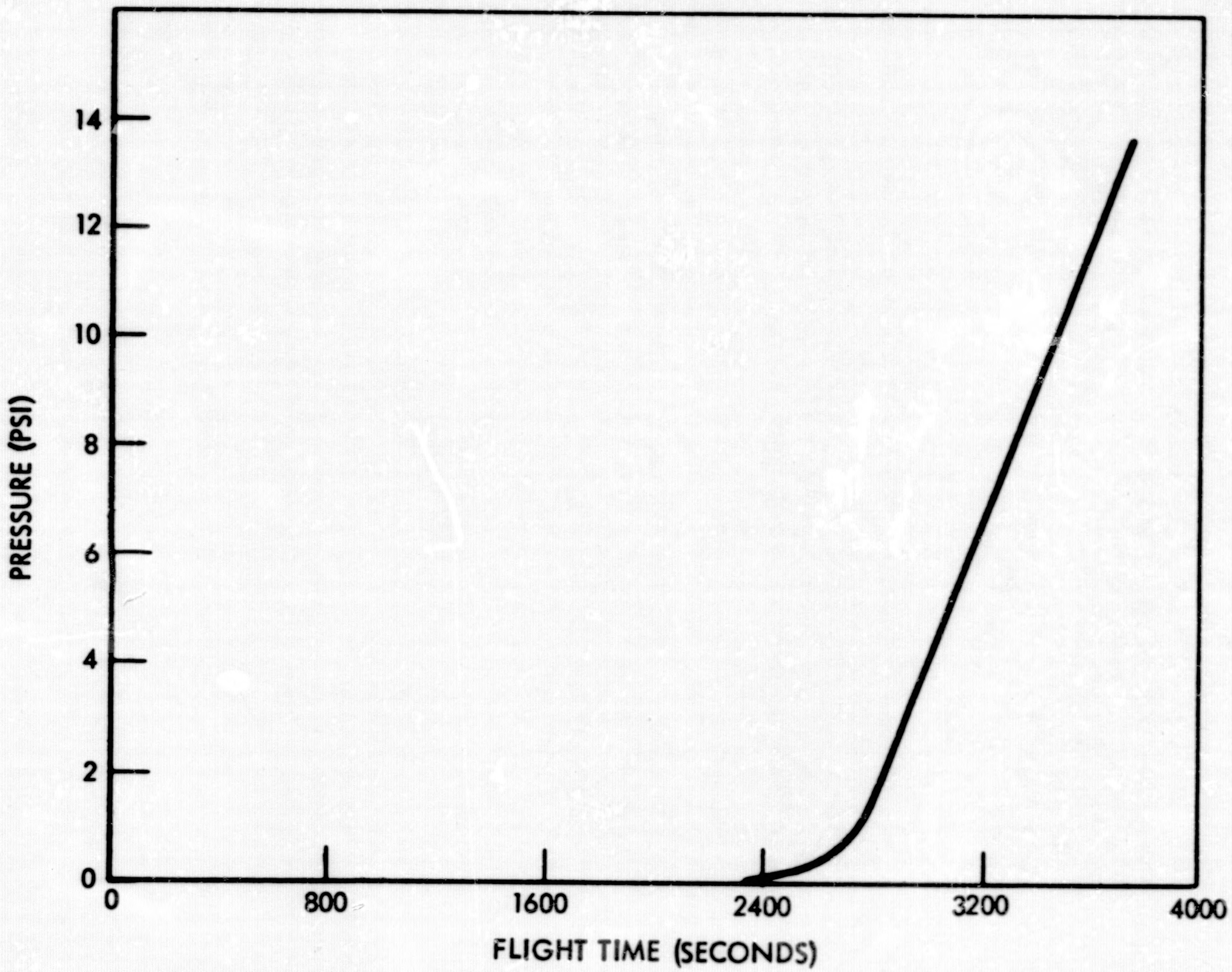


Figure 12.2-3 Cargo Bay Interval Pressure Time History During Re-entry

Using simple pressure-volume relations, with constant temperature, the leaking hydrogen will expand 760/2 or 380 times. The above leakage then expands to 380 times 0.25 or 95 chf. From Figure 12.2-3 the orbiter pressure reaches 2 mm Hg. in 2400 seconds or 2400/3600 hours, or a total leakage volume of 63 cu. ft. 63/16700 gives a concentration of 0.0038 or 0.38%.

As more precise information becomes available on (1) the free cargo bay volume, (2) the exact configuration of the cargo and its leakage, and (3) better methods to determine how much hydrogen remains in the cargo bay, the curve shown in Figure 12.2-4 would be more useful. This curve relates the total accumulated leakage to the free volume to arrive at a 4% hydrogen curve. The area above the curve would have greater than 4% H_2 and be hazardous and the area below would contain less than 4% and be safe.

Technical Discussion: Percentage Hydrogen in Hydrogen Tank Pressurant

The above discussion addressed itself to leaking hydrogen gas from many sources and as shown the leaking gas could be 100% hydrogen and not reach hazardous proportions under the conditions stated.

If a failure is assumed on the Tug when in the cargo bay after the shuttle has committed to re-enter and a "blowing" leak develops the leakage assumes a different character. Any leak from a point source will release the gas in jet form (Figure 3). This jet will interface with the atmosphere and at some point along this interface a reaction can occur if the percentage of hydrogen in the interface is 4% or greater. Reference (1) pointed out that hydrogen-oxygen reactions could occur when the percentage of oxygen in the mixed gases was 2% or greater. Since the oxygen source in the cargo bay will be the atmosphere and since air is approximately 20% or 1/5 oxygen, the interface mixture must contain a minimum of 5 x 2% or 10% air to provide enough oxygen to support a reaction. As a result 90% of the gas in the jet must contain enough hydrogen to bring the concentration of hydrogen to 4% in the interface mixture to permit a reaction. Therefore, the maximum allowable concentration of hydrogen in a blowing leak is 4%/90% or 4.5%. The gases escaping from a blowing leak will be diluted by the entrained air that provides the oxygen for a reaction and these gases must contain more than 4% hydrogen.

Since the leakage to be expected from components that are operating within specification is not hazardous, blowing leaks that are hazardous would be caused by a failure. If it is decided to protect the shuttle against these failures by inerting the hydrogen tank the hydrogen concentration must be $\leq 4.5\%$.

Technical Discussion: Energy Equivalent of a Blowing Leak of Hydrogen

As pointed out in reference (1) the intensity of the reaction hazard (fires or explosions) cannot be stated in a dynamic system. Concentration gradients would be set up from a blowing leak so that hydrogen/oxygen ratios are not relative if the source is above the minimum required to support a reaction.

The only effective control to minimize the hazard would be to leave the H_2 tank with 4.5% or better hydrogen and a blowing leak in orbit until repairs could be made. As shown in Figure 12.2-4 distributing leakage to be expected from well designed joints is not hazardous and leak rates great enough to be hazardous would be detectable by a tank decay test.

Even a shuttle or cargo mounted purge to inert the cargo bay would not eliminate the hazard since it would only serve to transfer the reaction to the cargo bay vents rather than the cargo bay itself. A flow-through air purge in the cargo bay would distribute the reaction from a blowing leak over a wider area and would be undesirable. A flow-through air purge to further dilute the hydrogen concentration from distributed leakage would be desirable, however.

Conclusions:

1. Normal leakage is not hazardous.

2. Blowing leaks are hazardous to the shuttle if the percentage of hydrogen is $\geq$4.5% (in the hydrogen tank).

3. Blowing leaks are detectable by visual examination, flight history and pressure decay checks.

4. Maximize the free cargo bay volume.

12.3 FINAL GUIDELINES AND REQUIREMENTS

Final Guidelines and Requirements for ground, ascent/descent, and mission operations are listed below and shall be incorporated in the Tug design. Safety criteria are categorized as they apply to all systems or individual systems.

All Systems

1. Inadvertent activation of critical systems shall be precluded through design considerations and the use of protective devices.

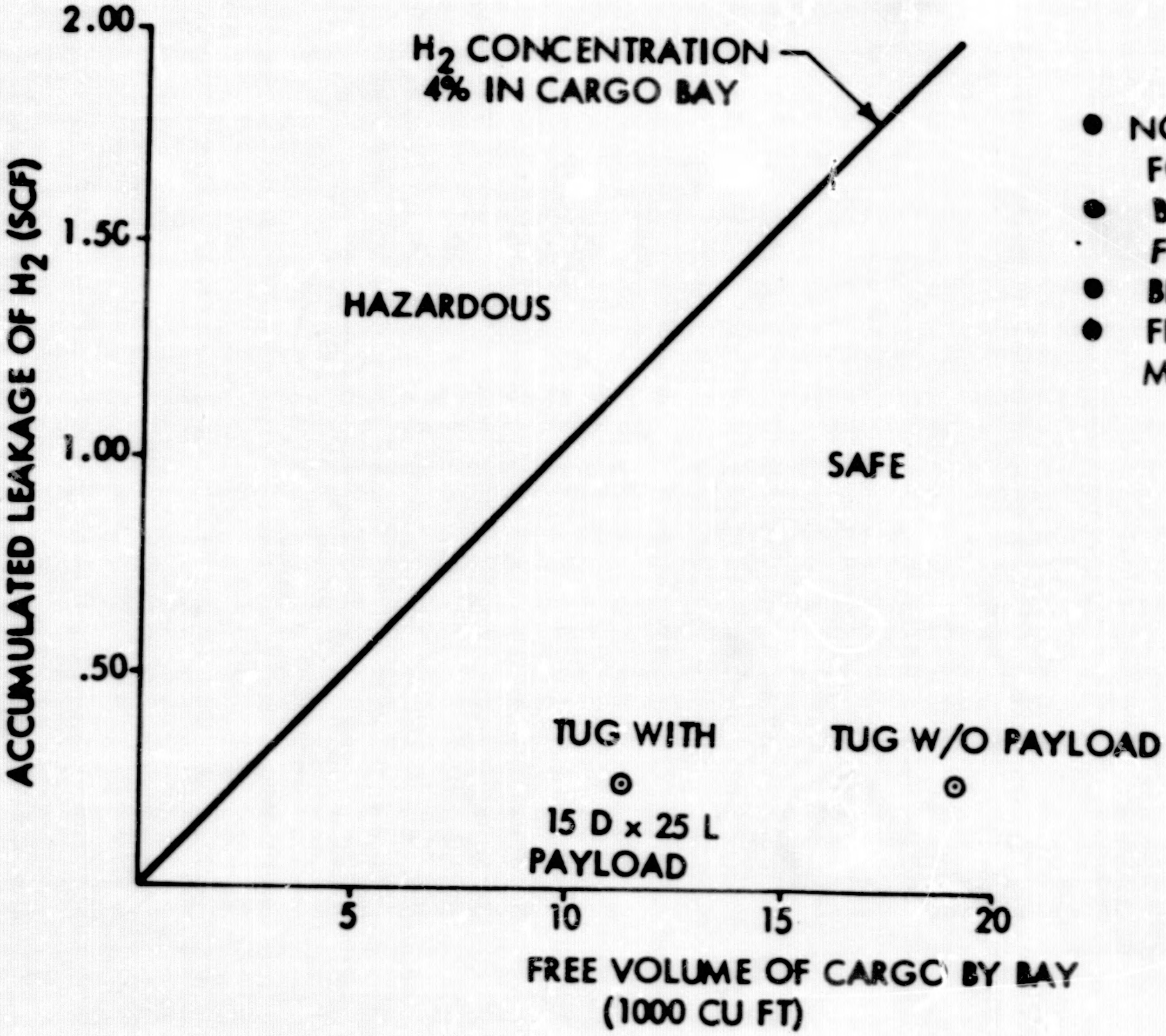
2. The capability shall be provided for performing critical functions at a nominal level with any single component failed or with any portion of a subsystem inactive for maintenance.

3. Provisions shall be made to eliminate metallic or non-metallic particles or parts, which could cause failure through lack of protection of the electrical equipment, in a zero "g" environment.

4. Provisions shall be made to protect equipment from damage or corrosion caused by electrolyte leakage from batteries.

5. The use of sulphur-containing or sulphur-coated materials in close proximity to electrical contacts is prohibited.

- 233 POSSIBLE LEAK POINTS ESTIMATED
- 0.25 SCF/H TOTAL LEAK RATE
- CUM LEAKAGE TO P = 2 mm Hg (2400 SEC)
- MOLECULAR DIFFUSION OF GASES



- NORMAL LEAKAGE IS SAFE FOR ANY CONC. IN LH_2 TANK
- BLOWING LEAK IS HAZARDOUS FOR LH_2 CONC $\geq$ 4.5%
- BLOWING LEAKS ARE DETECTABLE
- FREE VOLUME SHOULD BE MAXIMIZED IN THE DESIGN

Figure 12.2-4 System Safety Effort

6. Adjacent connections, electrical or fluid, should be configured differently to minimize cross connections.

7. Although the Tug is an unmanned vehicle, it will not compromise the man-rating of the orbiter while in or near the orbiter.

Propellant System

1. No shuttle cargo should be permitted to leak, vent, or discharge liquid propellants into the cargo bay.

2. Avoid venting bulk quantities of liquids if solid particles/chunks could impede subsequent operations.

3. The design of liquid dump lines must take into account that liquids discharging into a vacuum will evaporatively solidify and may restrict or block liquid flow.

4. A Tug which has sustained impact and penetration of propellant tanks by meteoroids or space debris should not be loaded into the cargo bay of the shuttle until it can be verified that no propellants remain.

5. The purge capability for inerting of integrated propellant systems will encompass the propellant system being inerted, from source through vent exhaust.

6. Automatic protective equipment will be incorporated in vehicle cryogenic systems to sense the onset of ullage pressure drop during the initial period of cryogenic loading, i.e., pressure level drops which could result in system failure (collapsing of tanks or tubing) due to cryogenics.

7. Cryogenic subsystems shall be designed to preclude the exposure of electrical/electronic components to cryogenic fluids.

8. Cryogenic fluid fill and drain systems shall be designed to minimize sensitivity to two-phase flow conditions and associated phenomena such as geysering.

9. Provisions shall be made for tank entry ports in propellant tankage for maintenance and inspection.

Pressurization System

1. Structures designed only for positive pressure will have provisions for vacuum relief and for the prevention of inadvertent depressurization.

2. Bypass circuits which override system pressurization interlocks are prohibited.

3. Positive design shall be provided to prevent reverse intallation of check valves.

Electrical System

1. Electrical components cooled to cryogenic temperatures shall be designed to preclude the ingestion of moisture and salts from external environments.

2. Leads of electronic parts shall be configured in a way that provides for relief of strain due to thermal expansion and contraction.

Structural System

1. Venting, drainage, and disposal provisions will be designed into structural compartments for the safe elimination of hazardous fluids.

2. Vent and drain openings at the vehicle mold line, including equipment drains, will be located such that drainage or vented gases will not re-enter the vehicle through other drain holes or vents.

3. All structural and mechanical systems will be designed with materials which are compatible with all working fluids and flushing and cleaning agents under the stress and environmental conditions of use.

4. Insulation will be compatible with the fluid within the propellant system to avoid hazards in the event of leaks.

12.4 CONCLUSIONS AND RECOMMENDATIONS

Conclusions

Normal system leakage is allowable.

Blowing leaks can be detected and precentive action taken before re-entry is initiated.

Blowing leaks are hazardous to the shuttle if the percentage of hydrogen by volume is greater than or equal to 4.5 percent.

Maximization of the free cargo bay volume minimizes hydrogen leakage hazards.

20-30 seconds after liftoff, the orbiter has no abort capabilities. Therefore, the tug has no propellant dump capabilities.

30-140 seconds after liftoff, the orbiter has two abort options: (1) fly back to launch site, or (2) fly to down range site. Flight time for both options is approximately 200-300 seconds, which is marginal to dump the maximum required weight of LOX to meet landing requirements.

140 plus seconds after liftoff, the orbiter is able to complete one orbit, re-enter, and land at the launch site or the down range site. Flight time is approximately 100 minutes, enough time to dump both LO_2 and LH_2 from the Tug, and perform inerting operations.

Recommendations

Conduct a detailed systems safety analysis.

Conduct further study to determine the Tug dump capabilities during abort. Consider time available for dump, line sizes for maximum dump, etc.

Conduct study to investigate the shuttle orbiter ability to land with 8,060-pound Tug payload.

As Tug systems are better defined, study the hydrogen leakage problem in more detail.

References:

12-1 Hydrogen-Oxygen Reactions Studies, Final Report, Dr. S. Kaye, GDC Report Number GDC-DBE-67-016, Contract NAS8-20350.

12-2 Tug Study Ground Rules

12-3 NR/SD Specification MA0620-001

12-4 Space Shuttle Orbiter Purge & Vent System, Study, Document SD71-169, August 1971, S. Kreidel

12-5 Safety in Earth Orbit, NR/SD Document SD71-569, Interim Draft of Final Report, Contract NAS9-12004, G. Canetti, October 1971

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VOLUME III

PART 2

APPENDIX A

FRACTURE MECHANICS ANALYSIS

FRACTURE MECHANICS ANALYSIS

The fracture mechanics analytical procedure for determining the necessary skin thickness for pressurized tanks is the same as that outlined in NASA SP-8040, "Fracture Control of Metallic Pressure Vessels." In general, the procedure requires that three material properties be obtained:

a. K_{IC} - Critical plane strain stress intensity

b. K_{TH} - Stress intensity at which a given flaw will continue to grow under a sustained load

c. da/dN - Flaw growth rate for different cyclic variations in stress intensified (ΔK)

Utilizing these properties, it is possible to determine the largest flaw which could exist in a pressurized vessel after a proof test; and the amount of flaw growth which would occur during the operational cycles, such that no flaw growth would occur under sustained load and the flaw could not grow through the thickness of the material.

The material properties indicated above are available for many materials, including 2219-T87 and 2014-T651 aluminums being considered for the TUG vehicle. However, when dealing with very thin materials (less than 0.2 inches) the commonly accepted values of K_{IC} and K_{TH} are no longer applicable. The thicknesses of material being considered for the TUG design are generally less than 0.1 inches.

The following discussion describes why the typical K_{IC}/K_{TH} values cannot be used for thin gages--and also provides a basis for development of the critical stress vs flaw size curves to be used for this study.

The normal depiction of K_{IC} in terms of critical stress and normalized flaw size is shown in Figure A-1. The solid line represents a constant value of K_{IC}, which is valid for thick materials (larger than 0.5 inches). However, for the space vehicle being analyzed, material thicknesses are generally less than 0.10 inches. For these thicknesses, only the dashed portion of the curve in Figure A-1 is applicable. The upper part applies as the stress approaches the yield strength of the material, and the low end of the curve applies when the flaw approaches the far surface of the metal. The resulting curve looks like that shown in Figure A-2.

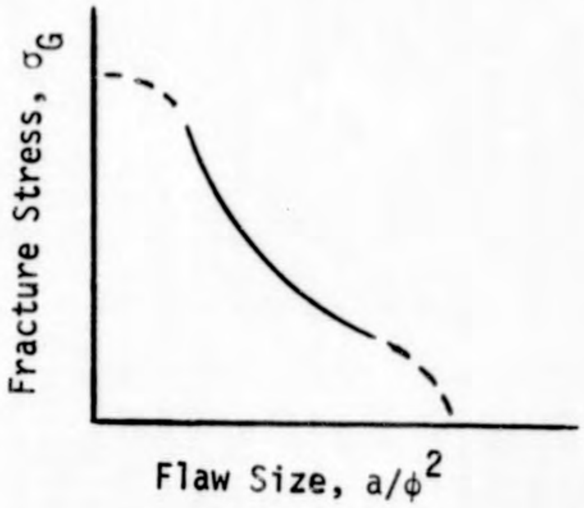


FIGURE A-1

In this range, linear elastic fracture mechanics does not apply. The reasons being that (1) the flaw is exposed to a plastic straining at the yield strength of the materials, and (2) the remaining ligament between the crack front and the far surface becomes so small that it can no longer carry the load and hence, begins to deform plastically. Figure A-2 represents the effect that these two phenomena have on the critical fracture stress for various flaw sizes in one thickness of material. Other thicknesses of material must be represented by different curves, so that a family of curves (Figure A-3) is required to represent the critical stress for a wide range of flaw sizes. These are the type of curves to be used in selecting the critical flaw size which would be screened by a proof test.

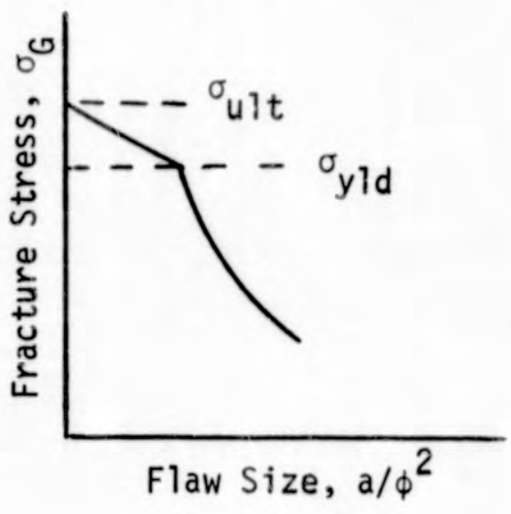


FIGURE A-2

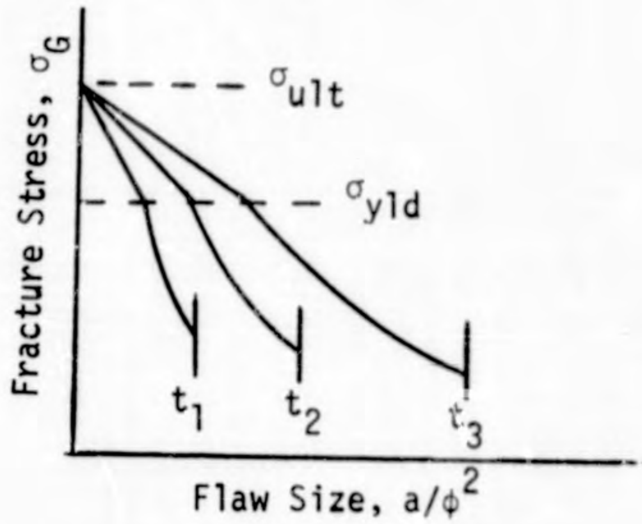


FIGURE A-3

The following section identifies the material properties data needed to perform a fracture mechanics analysis of the TUG vehicle.

FRACTURE MECHANICS MATERIAL DATA

The fracture mechanics data utilized in this report was either: extracted from reports by other investigators; obtained from current NR testing programs; developed by extrapolation of data; or estimated based on analysis and observed material characteristics. The material properties selected include: tensile strength at different temperatures; critical fracture stress vs flaw size at RT and -320 F; critical stress for flaw growth vs flaw size at RT and -320 F; flaw growth rate vs cyclic stress intensity (ΔK) at RT and -320 F. Material strength properties for 2219-T87 and 2014-T651 are summarized in Table A-1.

Critical Fracture Stress vs Flaw Size

There is very little data available which can be used to develop the Critical Fracture Stress vs Flaw Size curves. Therefore, it is necessary to develop these curves utilizing existing data, knowledge of material fracture characteristics, and linear elastic fracture mechanics analysis.

Table A-1. Tensile Strength

Material		F_{tu}			F_{ty}		
		RT	-297	-423	RT	-297	-423
2219-T87	(L)	63	75.6	91.3	52	60.8	69.1
	(T)	64	76.8	92.8	52	60.8	69.1
	(*ST)	60.0	72.0	87.0	46.0	53.8	61.1
2014-T657	(L)	67	76.3	90.4	59	67.2	75.5
	(T)	66	75.2	89.1	58	66.1	74.2
	(**ST)	60	68.4	81.0	54	61.5	69.1

MIL HDBK-5 (.040 - .249 thickness)
*Values are for hand forgings, not sheet and plate
**Values are for 2.0 inch plate

As mentioned in the introductory comments, the expected fracture stress curve will look similar to that shown in Figure A-4. The procedure to be followed in developing this curve is based on the fact that the strength of the material affects the high end of the curve, and net ligament thickness and fracture toughness affects the low end of the curve. The general steps which were followed in developing the curve were:

1. Obtain fracture data for flawed specimens which failed at or above gross section yield.

2. Establish a general pattern of expected fracture results for other thicknesses.

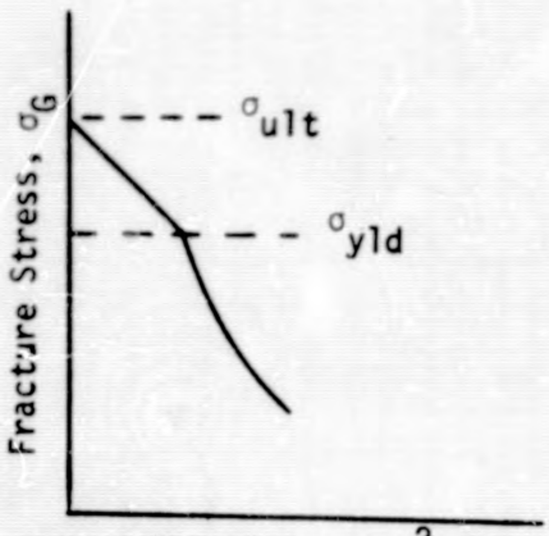


FIGURE A-4

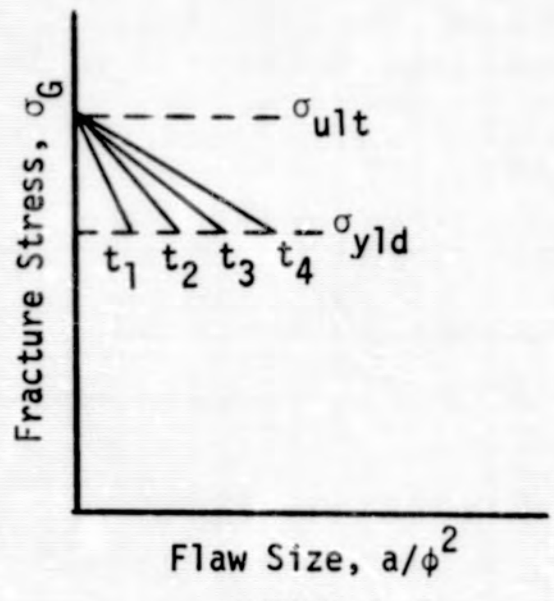


FIGURE A-5

These two steps will provide a depiction of the upper portion of the curve for several thicknesses. The intersection of these partial curves, with the dashed line representing the yield stress, is where fracture toughness begins to exert significant influence. The next steps followed in developing the curve considered the fracture toughness influence:

3. Determine apparent fracture toughness variation with respect to a/ϕ^2

   NOTE: In the thickness of material being considered, two elements affect fracture toughness: thickness of material, and flaw size. These effects are shown in Figure A-6.

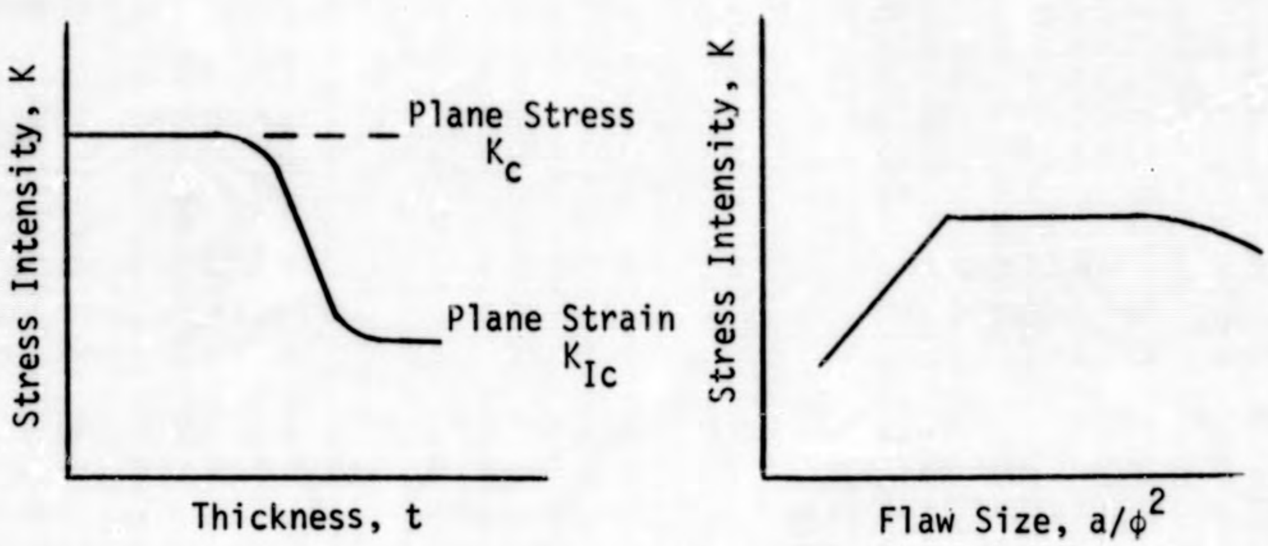


FIGURE A-6

4. Calculate and plot the critical stress for different a/ϕ^2 values considering different thicknesses and using the Kobayashi far surface correction factor up to a flaw depth to thickness ratio (a/t) of 0.8.

   NOTE: Theories which attempt to correct for premature failure of test specimens, as the flaw approaches the far surface of the metal, do not work well beyond an a/t of 0.8. For instance, investigators using the Paris for surface correction factor will find it approaching infinity as the flaw approaches a/t = 1. The Kobayashi M_K far surface correction factor is generally faired into a value of 1.6 as the flaw approaches the far surface.

5. Extend lines which are tangent to each of the curves, developed in 4, above, from a flaw depth to thickness ratio of 0.8 to the respective far surfaces.

Step one of the curve development procedure required a review of available data to determine fracture data from 2219-T87 and 2014-T651 aluminum test specimens which contained part thru crack type flaws and fractured above the yield strength of the material. This data is summarized in Figures A-7 and A-8.

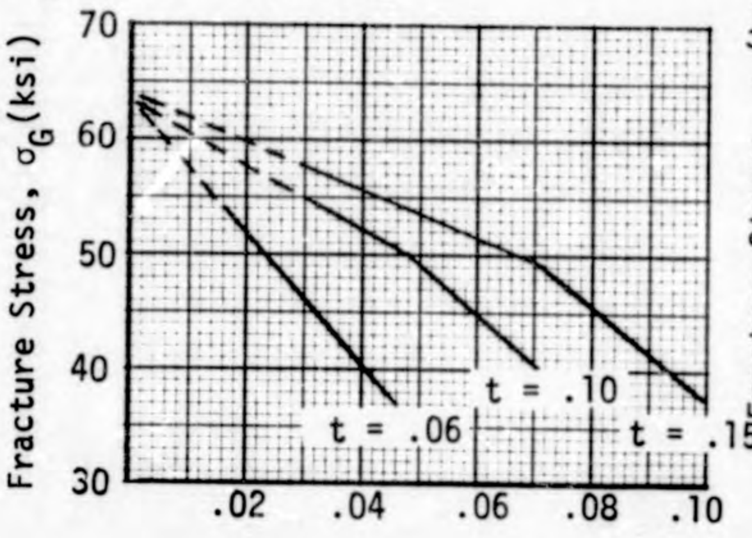


FIGURE A-7

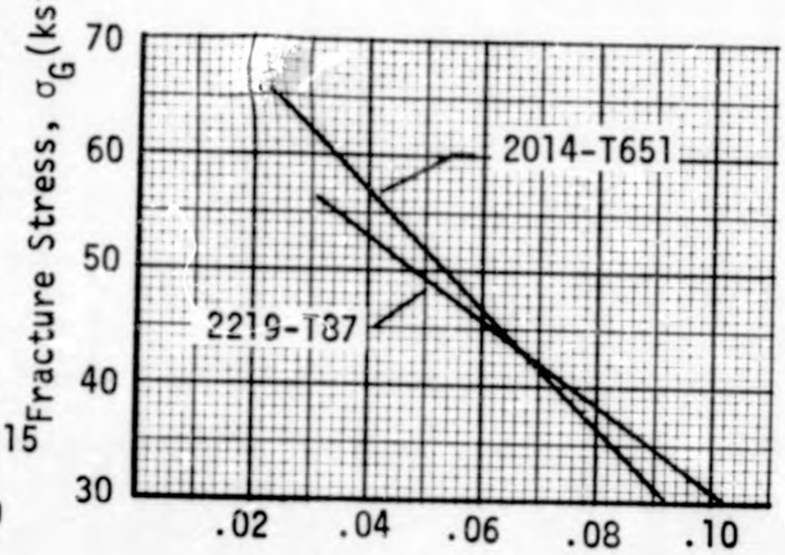


FIGURE A-8

Figure A-7 represents 2219-T87 fracture strength where the solid lines are test data and the dashed lines extrapolations of the data. The test results were obtained from current NR laboratory investigations into the behavior of 2219-T87. The results of the investigation have shown that when gross section yielding is required to fracture a test specimen, the relationship between flaw size and failure stress becomes nearly linear. This is shown by the dashed and solid lines of Figure A-7, and is the information used to depict the upper portion of the Fracture Stress curve.

Figure A-8 shows a comparison of 2219-T87 and 2014-T651 data in the range of thicknesses being considered in the analysis. It can be noted that the apparent toughness of the 2014-T657 is exceeding the toughness of the 2219-T87 as the flaw size approaches zero. This is contrary to what would be concluded by comparing K_{IC} values, and occurs because the yield and ultimate strengths of 2014-T651 are higher than the 2219-T87 strengths. In the small flaw size range, strength is the predominant factor in determining material toughness. This fact allows the development of the upper part of the 2014-T651 Fracture Stress curve.

Low temperatures will cause the strength of the materials to increase and as a result cause a similar change in the Fracture Stress curve. A parallel shift of the Fracture Stress curve to account for temperature would look like the solid line A-B shown in Figure A-9. However the intersection of the low temperature curve with the -300° yield strength line is dependent on the apparent fracture toughness of the material and would be represented by the dashed line A-C. The dashed line C-D is based on the apparent

fracture toughness of the material. Point C can be estimated by extending the curve E-F up to the -300°F yield strength line and then verified when the lower part of the Fracture Stress curve has been determined.

Utilizing the material properties obtained and the assumptions discussed above it is now possible to provide an estimate of the Fracture Stress properties of 2014-T651 and 2219-T87 at R.T., -300°F and -423°F between the respective yield and ultimate strengths. This information is summarized in Figure A-10.

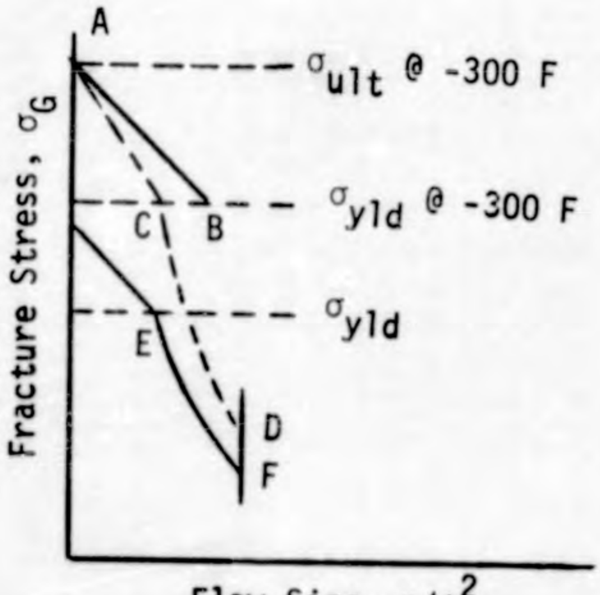


FIGURE A-9

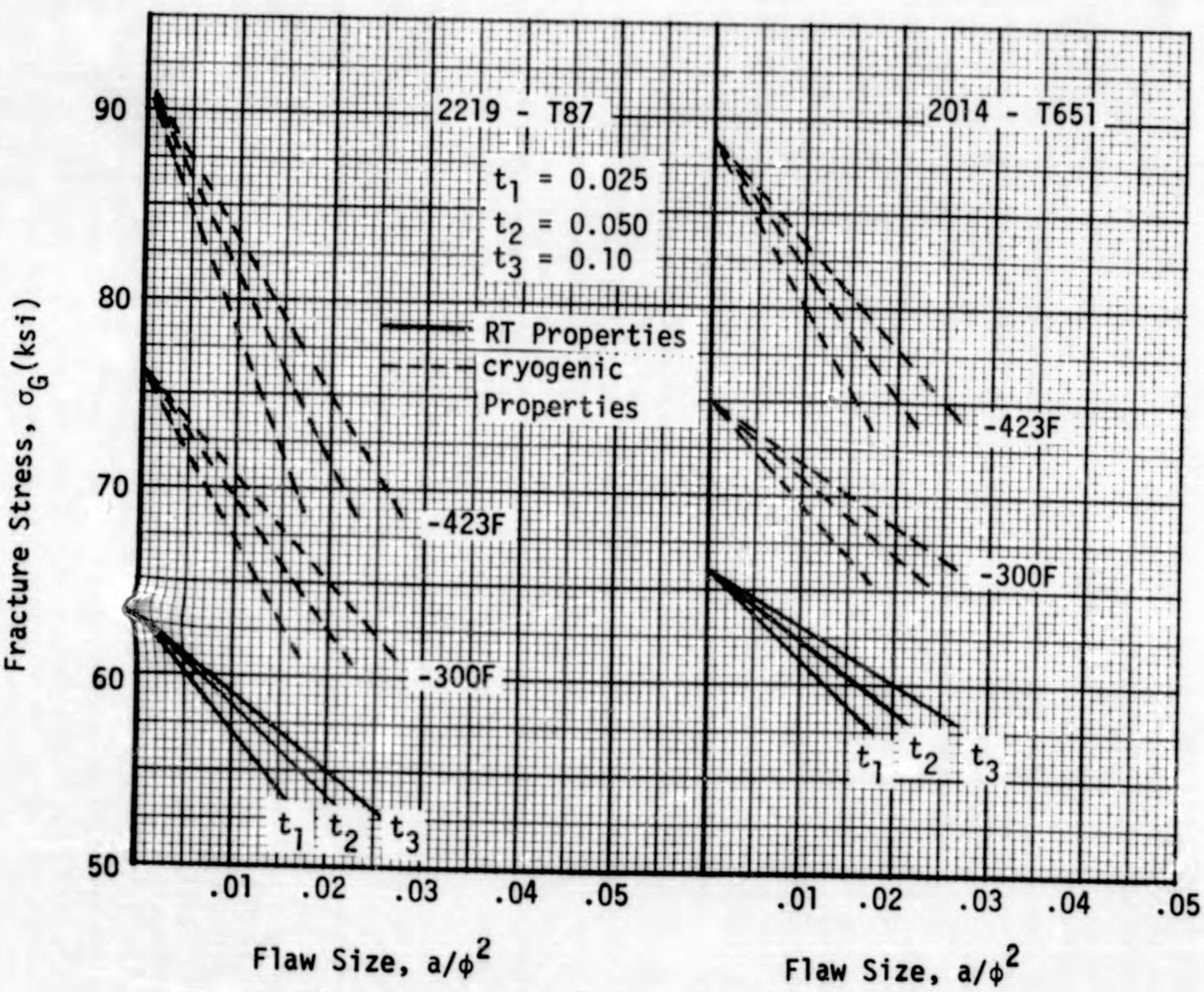


FIGURE A-10

Step 3 of the Fracture Stress curve development requires that the apparent stress intensity vs a/ϕ^2 curve be established. The curve will be established for a 70°F environment and then corrected as required for low temperature environments.

Reviewing Figure A-7 it is noted that 2219-T87 test information is available which allows determination of an apparent stress intensity for small values of a/ϕ^2 in the critical stress range of 40,000 psi to 50,000 psi. The critical fracture stresses in the vicinity of the material yield strength combined with low (a/ϕ^2)'s allows for the determination of the lower bound values of fracture toughness and is shown by the shaded line in Figure A-11.

The results of the analysis are shown in Figure A-12.

These curves are primarily dependent on fracture toughness of the material. Since the values of K_{IC} for both 2219-T87 and 2014-T651 are nearly the same (Reference Figure A-8), it will be assumed that the above curves are representative of both materials. The RT curve was established from raw test data and the low temperature

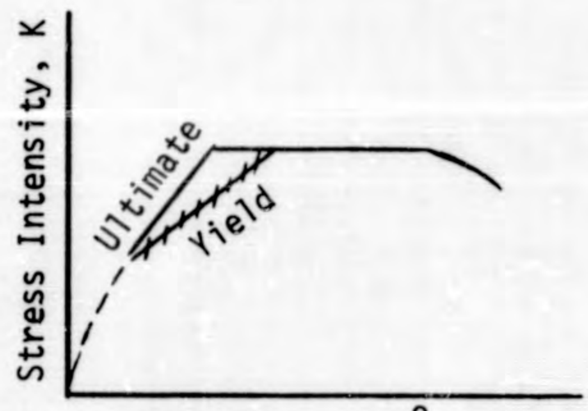


FIGURE A-11

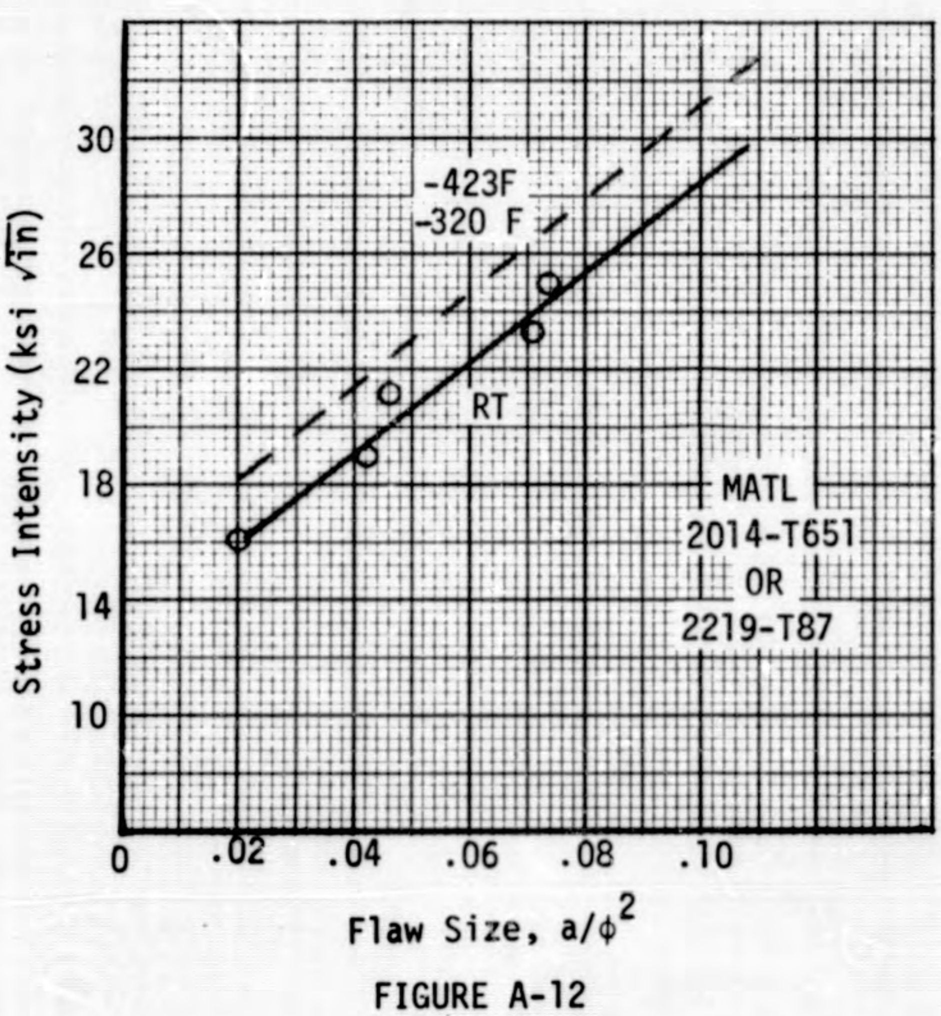


FIGURE A-12

curve established by comparing published values of K_{IC} at RT, -300°F and -423°F. It was determined that little or no difference in K_{IC} existed between -300°F and -423°F for either material. However the low temperature values of K_{IC} are approximately 10% greater than the values determined at room temperature. This low temperature correction of the critical stress intensity is reflected by the dashed line in Figure A-12. It should be noted that the values of critical stress intensity shown in Figure A-12 are independent of thickness but dependent on flaw size. The next step taken will provide the thickness influence on the critical stress at fracture.

Step 4 requires evaluation of the fracture stress for various thicknesses and flaw sizes (a/ϕ^2). The equation used to obtain the fracture stress was:

$$K = \left[1 + .12 \ (1 - \frac{a}{c}) \right] \sigma \sqrt{\frac{\pi a}{Q}} \ M_K$$

where

K = Stress intensity at fracture

a = Flaw depth

c = One-half the flaw length

σ = Stress at fracture

Q = Flaw normalizer with plasticity correction

$= \phi^2 - 0.212 \left(\frac{\sigma}{\sigma_y}\right)^2$

$$\phi = \int_o^{\pi/2} \left(1 - \frac{c^2 - a^2}{c^2} \sin^2\theta\right)^{1/2} d\theta$$

σ_y = Yield stress

M_K = Kobayashi far surface correction factor

The far surface correction factor used is shown in Figure A-13 and is the factor which is affected by flaw size and material thickness.

Results of the analysis are shown in Figures A-14 and A-15 which also include the results obtained in step two of this design curve development procedure.

Step 5 required extending lines which are tangent to each of the curves developed in Step 4, from a flaw depth to thickness ratio of 0.8 to the respective far surfaces. This was accomplished by completing the design curves depicting the critical Fracture Stress vs Flaw Size. The completed curves are shown in Figures A-14 and A-15.

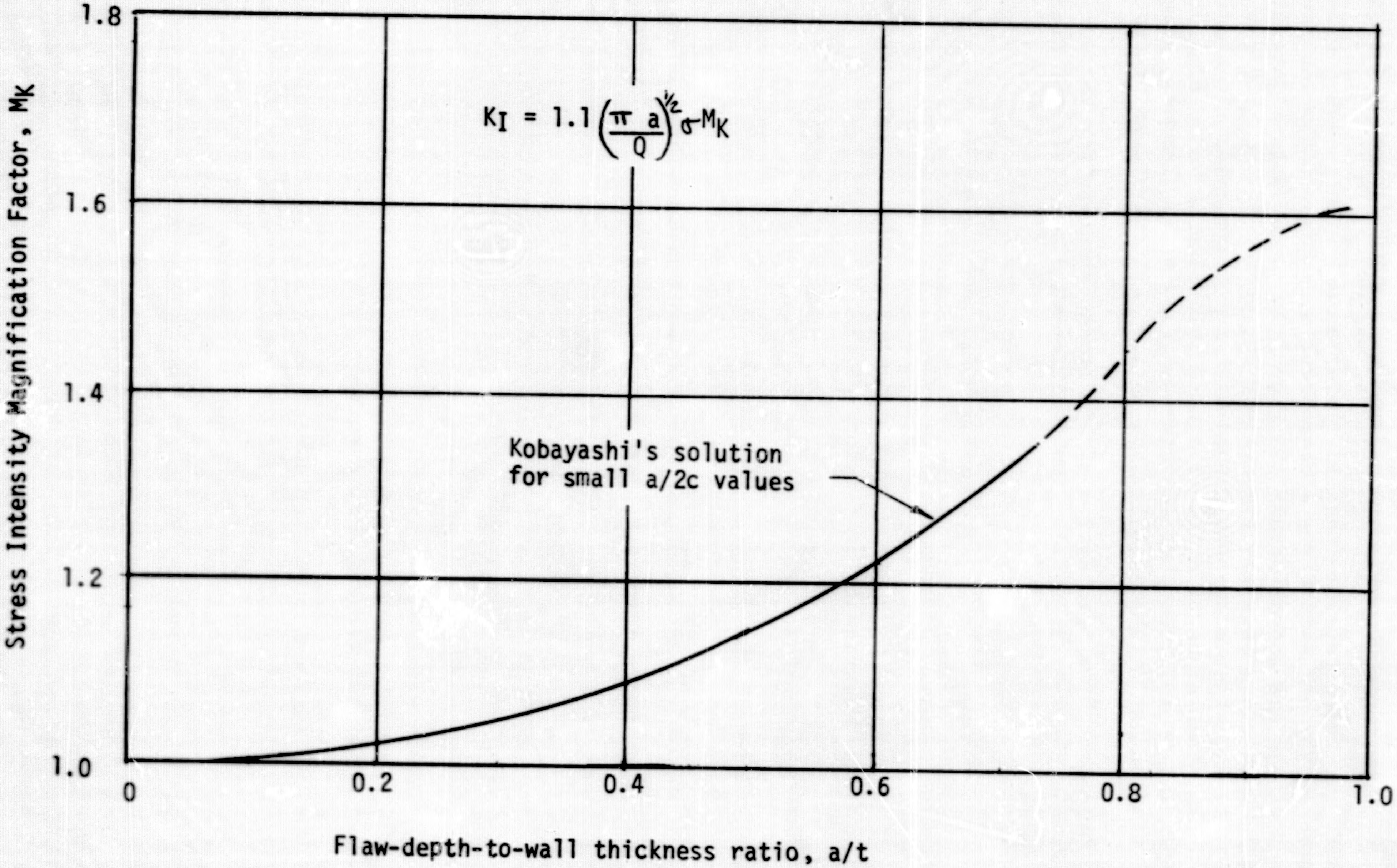

1.8
1.6
1.4
1.2
1.0
0
0.2
0.4
0.6
0.8
1.0
Stress Intensity Magnification Factor, M_K
$K_I = 1.1\left(\frac{\pi a}{Q}\right)^{1/2}\sigma M_K$
Kobayashi's solution
for small a/2c values
Flaw-depth-to-wall thickness ratio, a/t

FIGURE A-13

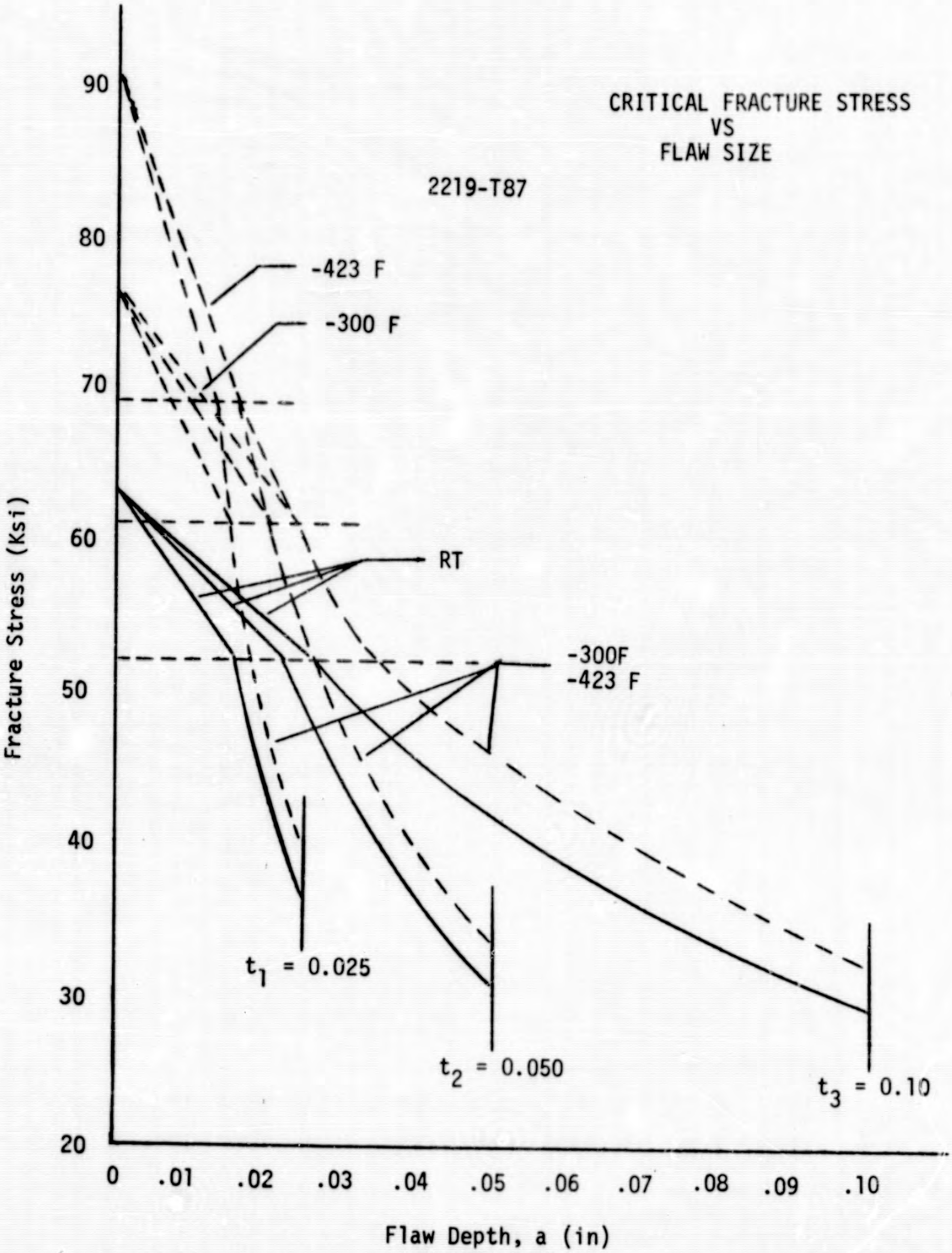

CRITICAL FRACTURE STRESS
VS
FLAW SIZE
2219-T87
-423 F
-300 F
RT
-300F
-423 F
t_1 = 0.025
t_2 = 0.050
t_3 = 0.10
Fracture Stress (Ksi)
90
80
70
60
50
40
30
20
0
.01
.02
.03
.04
.05
.06
.07
.08
.09
.10
Flaw Depth, a (in)

FIGURE A-14

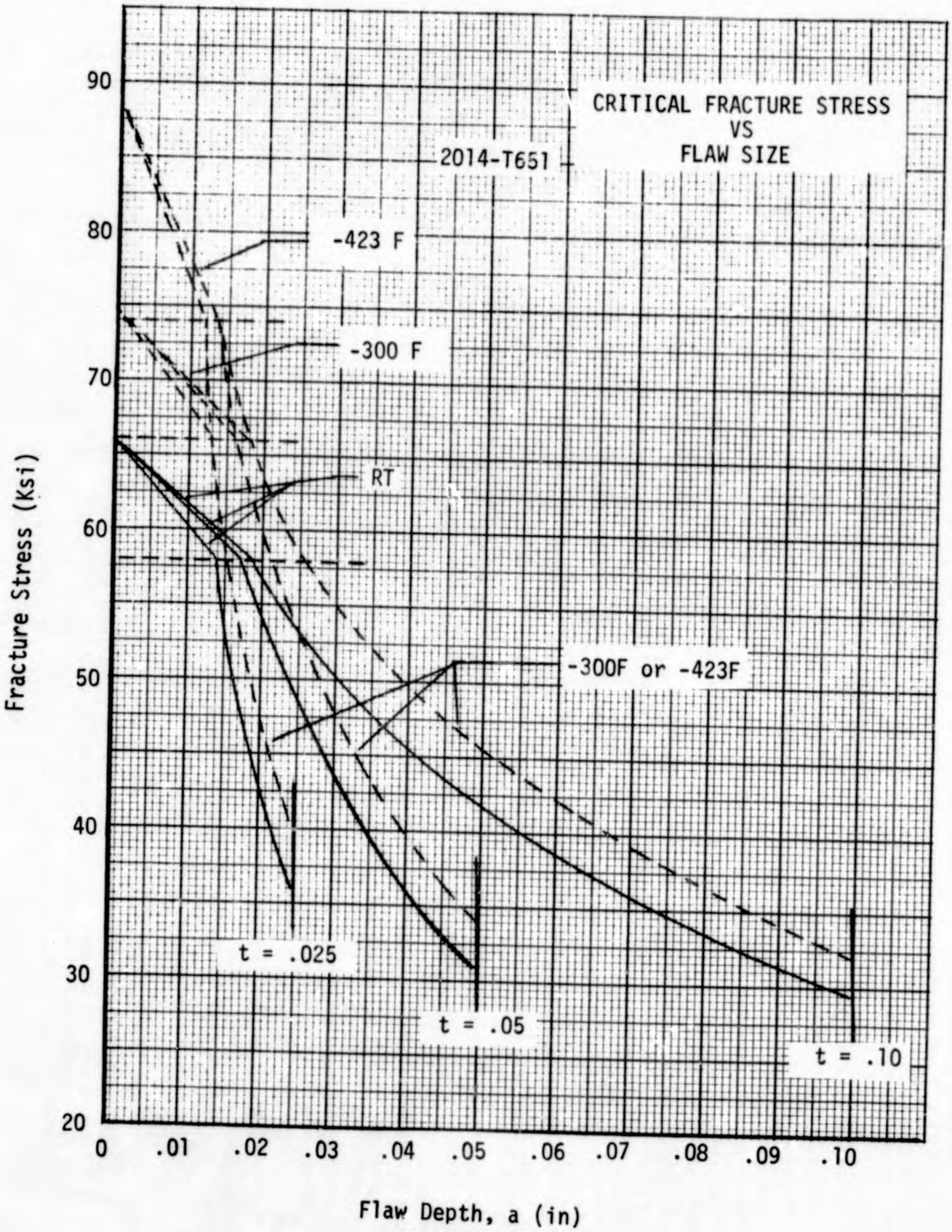

CRITICAL FRACTURE STRESS
VS
FLAW SIZE
2014-T651
-423 F
-300 F
RT
-300F or -423F
t = .025
t = .05
t = .10
Fracture Stress (Ksi)
90
80
70
60
50
40
30
20
0
.01
.02
.03
.04
.05
.06
.07
.08
.09
.10
Flaw Depth, a (in)

FIGURE A-15

The next set of design curves required designate the stress which can be sustained with no coincident flaw growth.

Sustained Stress Threshold Vs Flaw Size

An indication of the sustained stress threshold for 2219-T87 is given in SP 8040 where ratios of K_{TH}/K_{IC} are provided for RT, -300°F and -423°F. These values are:

Temperature °F	K_{TH}/K_{IC}
RT	0.90
-320	0.82
-423	0.85

Testing has been performed at NR which indicates that the R.T. sustained stress threshold is in excess of 97% of K_{IC} when exposed to a laboratory environment. This is a significant difference from the value of 90% listed in SP 8040. There may be one or more reasons for this variation in test results, e.g., the values listed in SP 8040 may represent stress at which flaw growth can occur but not the stress which is required to maintain flaw growth. The testing performed by NR provided for continuous monitoring of the flaw size. Typical results indicated that the flaw would grow on the initial load application then arrest with no subsequent growth indicated.

As a result of the above observations it will be assumed that K_{TH} = .95 K_c at room temperature and K_{TH} = 0.90 K_c at both -300 and -423°F. The value of 90% K_c would be a more accurate depiction of K_{TH} at -320°F. This means that the K_{TH} = 0.90 K_c at -423°F is a conservative estimate.

Since no information was found which would indicate the K_{TH} for 2014-T651 it was assumed that the relationship between K_c and K_{TH} would be the same as for the 2219-T87. The values of K_{TH} are shown on Figure A-16.

Cyclic load flaw growth data is the remaining information needed to perform a fracture mechanics analysis. This information is developed and discussed in the following paragraphs.

Cyclic Stress Intensity (ΔK) vs Flaw Growth Rate (da/dN)

Flaw growth data for 2219-T87 was obtained from a NASA-MSC report, MTBR71-ES5-1, "Fatigue Flaw Growth Behavior of 2219-T87 Aluminum at Cryogenic, Room and Elevated Temperatures," dated Sept. 2, 1971. The data obtained is shown in Figures A-17 and A-18. Empirical material constants shown on the figures are utilized in the growth rate equation shown below. However these constants represent values obtained from a best fit of the test data. An alternate dashed line representing flaw growth is drawn to establish a lower bound of the test data. The empirical constants used to represent the alternate flaw growth rate are shown below the dashed line.

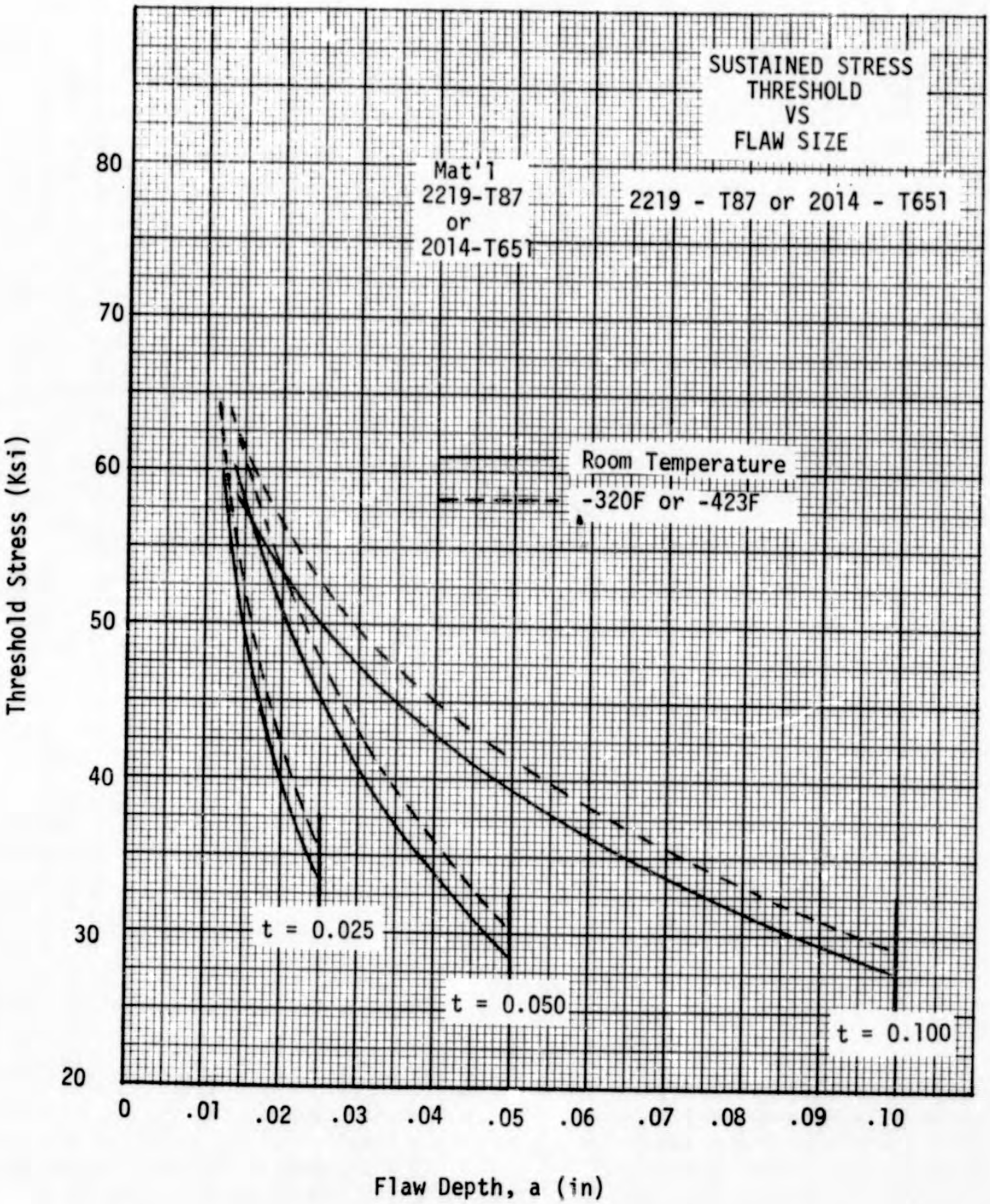

SUSTAINED STRESS
THRESHOLD
VS
FLAW SIZE
Mat'l
2219-T87
or
2014-T651
2219 - T87 or 2014 - T651
Room Temperature
-320F or -423F
t = 0.025
t = 0.050
t = 0.100
Threshold Stress (Ksi)
20
30
40
50
60
70
80
0
.01
.02
.03
.04
.05
.06
.07
.08
.09
.10
Flaw Depth, a (in)

FIGURE A-16

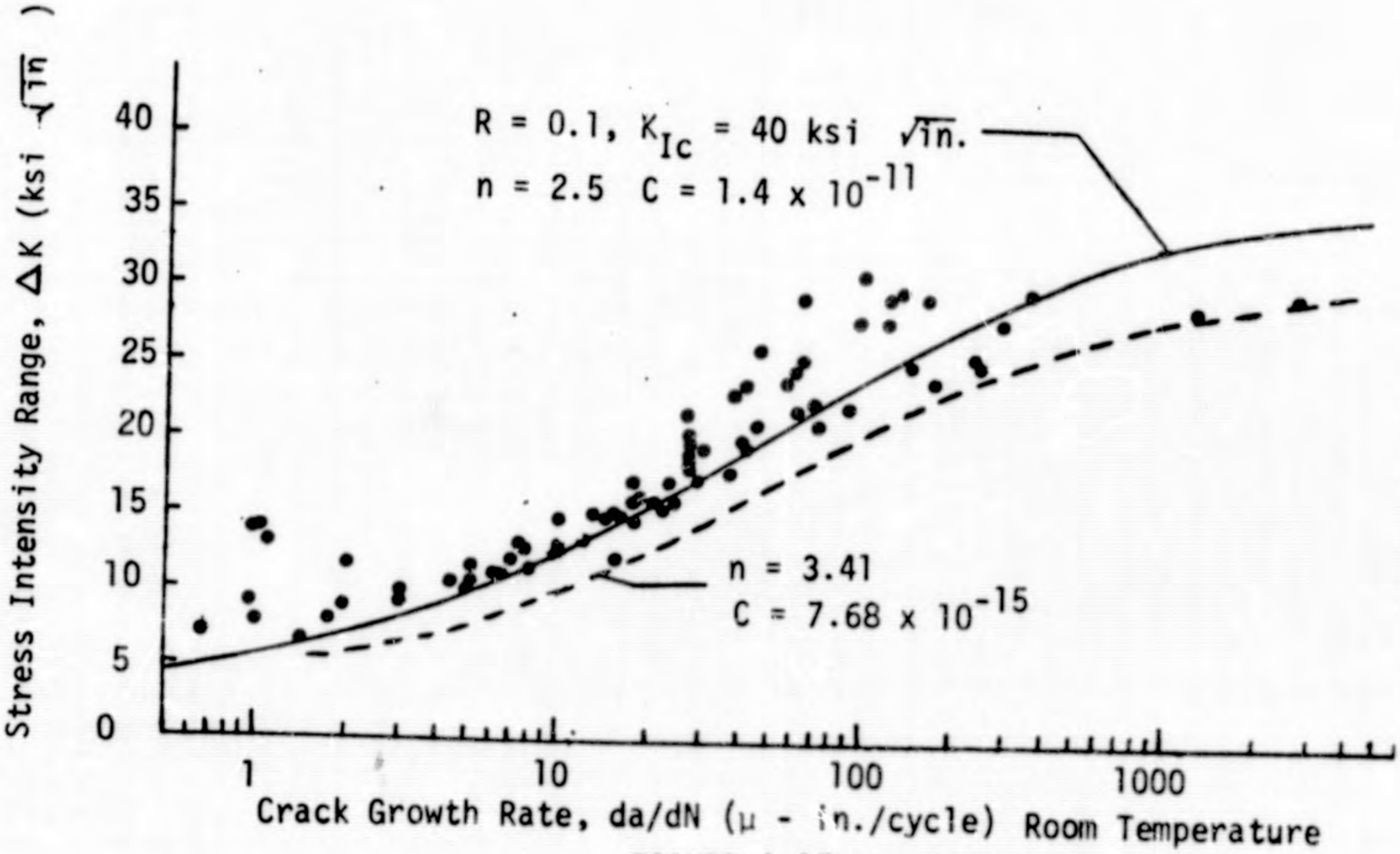


FIGURE A-17

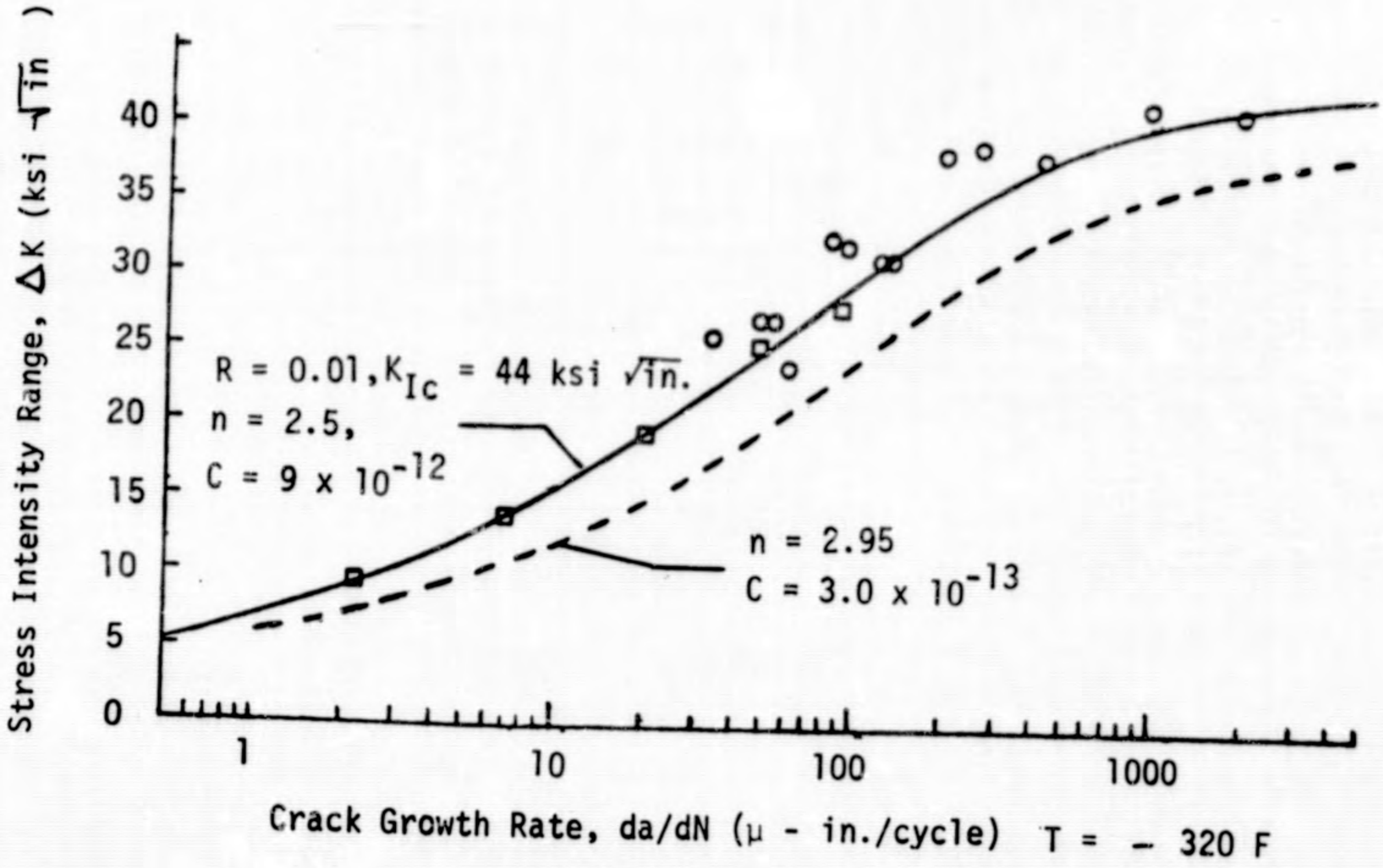


FIGURE A-18

$$\frac{da}{dN} = \frac{C\ (\Delta K)^n}{(1 - R)\ K_{IC} - \Delta K}$$

where

$\frac{da}{dN}$ = Flaw growth rate

C and n = Empirical constants

R = Stress ratio $\frac{\sigma\ Min}{\sigma\ Max}$

K_{Ic} = Critical stress intensity

$K = [1 + .12\ (1 - \frac{a}{c})]\ \sigma \sqrt{\frac{\pi a}{Q}}\ M_K$

ΔK = Change in stress intensity

Flaw growth rate information was not available for a temperature of -423°F. Since there is little difference in the critical stress intensity between -300°F and -423°F for either 2219-T87 or 2014-T651, it was assumed that the flaw growth rates were unchanged between -300°F and -423°F.

Flaw growth data for 2014-T651 was not available for the analysis, therefore, it was necessary to use the 2219-T87 growth data for analysis of flaw in 2014-T651 pressure vessels.

Analysis

Using the data presented in the preceding section, it is possible to select the propellant tank material and to estimate the tank wall thicknesses required to survive a proof test and assure a vehicle life of 20 mission cycles. It should be noted that no fracture mechanics evaluation of welds is included in the analysis. The reason welds were not considered is that the weight of the propellant tanks is based primarily on the thickness of the parent material and that weld land thickness variations required by fracture mechanics analysis will have an insignificant impact on the final structural weight.

The following discussion describes the assumption and rationale used in selecting the material:

Material Selection

The objective in the material selection process is to determine which material can be the thinnest gage and still survive the proof test and 20 mission cycles. This requires knowledge of the three material characteristics determined in the previous section, Fracture Stress vs Flaw Size, Sustained Stress Threshold vs Flaw Size, and Cyclic Stress Intensity vs Flaw Growth Rate. In the preceding sections, it was assumed that the Sustained

Threshold vs Flaw Size and the Flaw Growth Rates were the same for 2014-T651 and 2219-T87. This reduces the selection problem to that of determining which material allows screening of the smallest flaw by a proof test. Screening the smaller critical flaw allows a greater remaining thickness of material for flaw growth. The alternate to more material being available for flaw growth is that of reducing material thickness so that the flaw size to thickness ratio is the same in both materials being considered. Hence, assuming flaw growth rates and sustained stress flaw growth thresholds are the same, the smaller critical flaw screened by a proof test allows selection of the thinner material.

A portion of Figures A-14 and A-15 are shown below, so that a comparative analysis can be made to determine which material, when subjected to proof test, will screen out the smallest flaw. It will be assumed that the proof test performed will limit the maximum stress to 95% of the yield stress of the material.

Comparison of the two sets of curves can be accomplished by evaluating an example. Consider the RT Fracture Stress curve depicting the 0.05 thick material for both 2014-T651 and 2219-T87. The 2219-T87 curve (Figure A-19a) shows that at 49,400 psi (0.95 F_{ty}) the flaw screened by a proof test is 0.0243. Figure A-19b (2014-T651) shows a flaw size of 0.0195 is screened by a proof test of 55,100 psi (0.95 F_{ty}). This indicates that the 2014-T651 is the better material. A review of the remaining curves on these two figures provides the same conclusion. This is the material which will be used in the remaining flaw growth analysis.

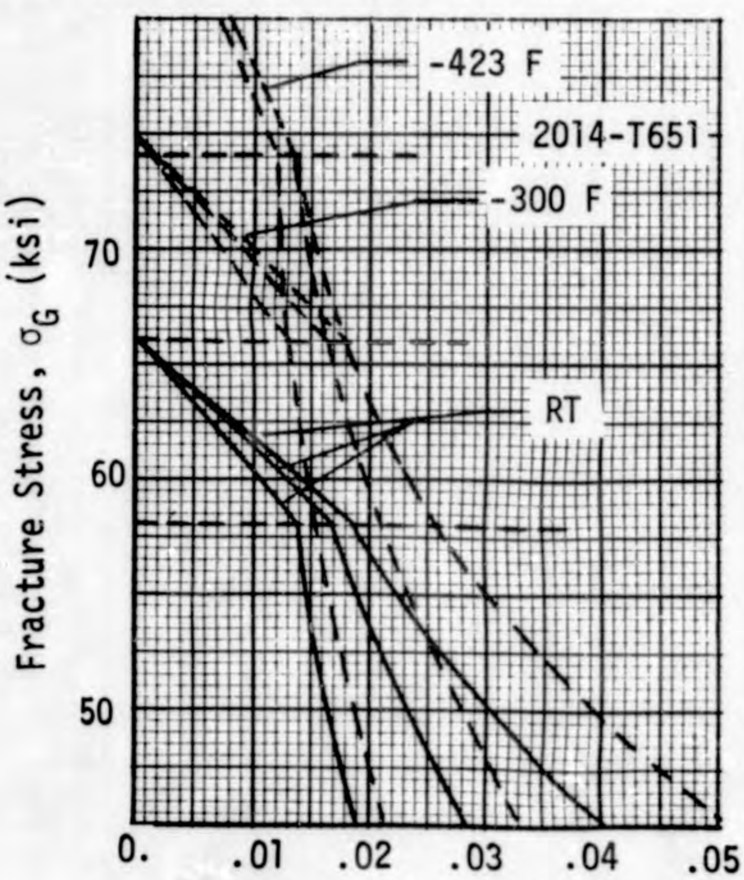


FIGURE A-19a

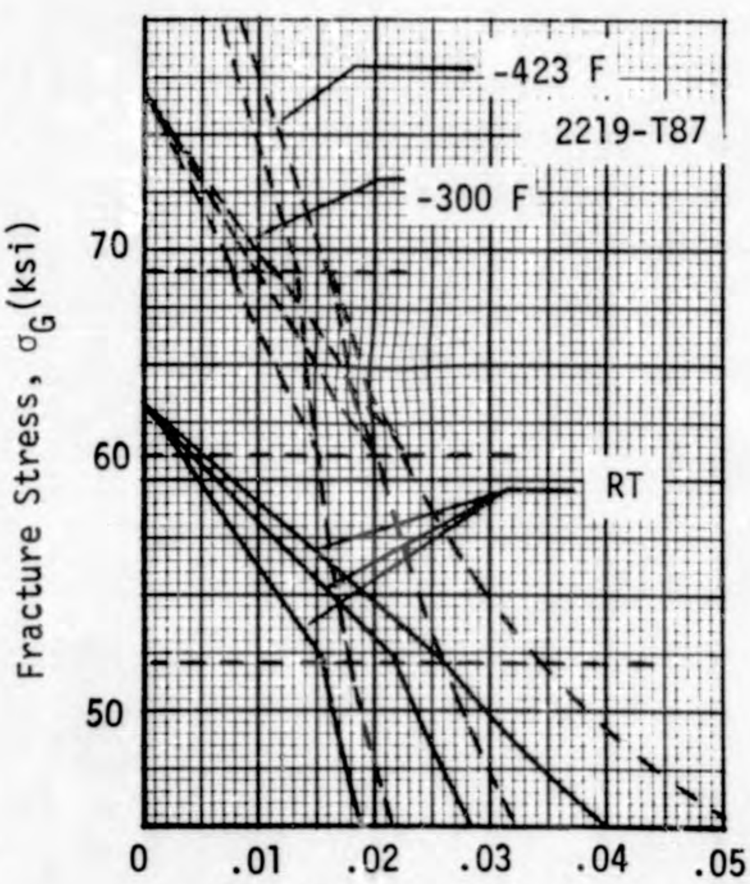


FIGURE A-19b

As previously noted, there are a number of assumptions which allowed development of the material properties used in this analysis. Material selection is dependent on the validity of the assumptions; therefore, final selection of material should not be made before obtaining detailed test information for both materials considered, which verifies or contradicts the assumptions and, hence, the material selection made by the above analysis.

With the completion of the material selection process and having identified the material characteristics, it is now possible to determine the desirable proof test; maximum flaw which could exist in the pressure vessels after a proof test; and the amount of flaw growth which would occur during the operational cycles.

Flaw Growth Analysis

A flow chart depicting the fracture mechanics evaluation procedure is shown in Figure A-20.

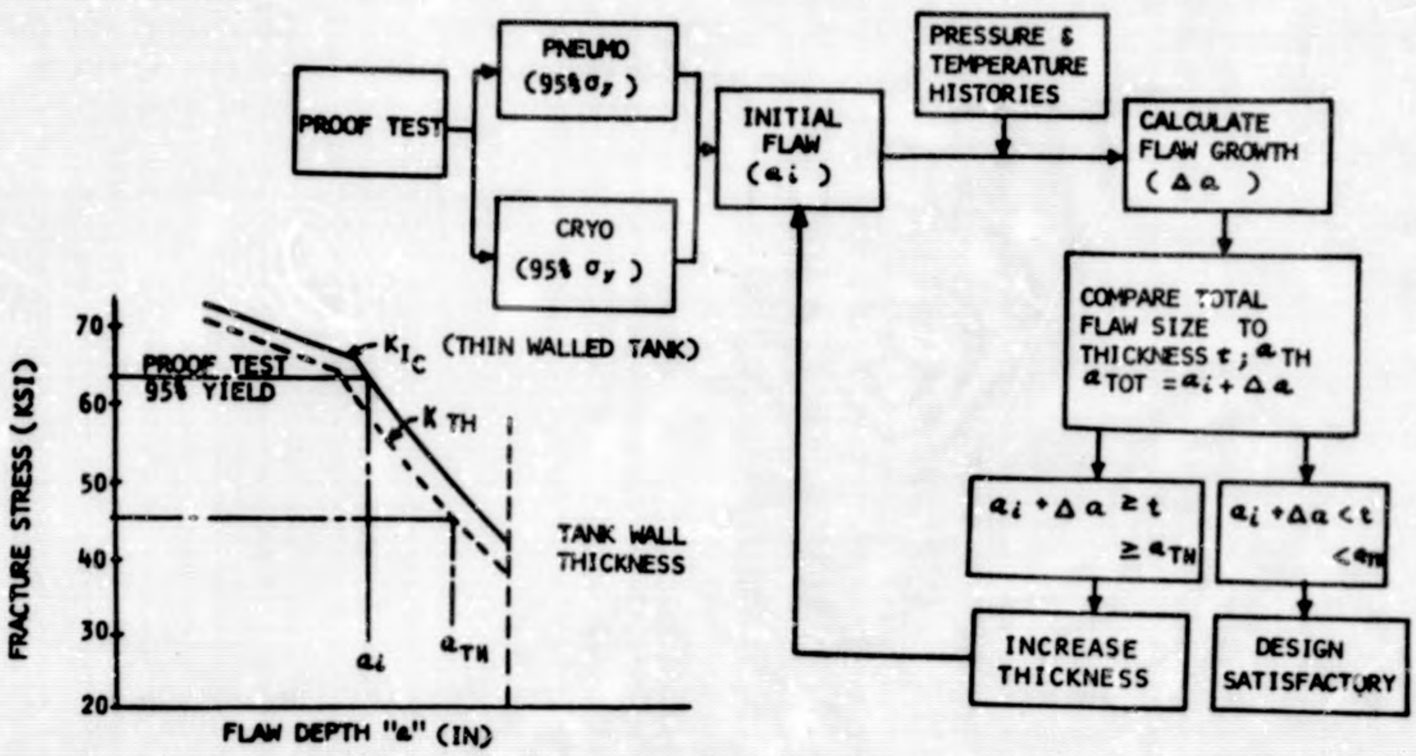


FIGURE A-20

Selection of a proof test required consideration be given to test temperature, maximum stress level test fluid, inspectability of flaws screened by the proof test, and potential failures which could occur during the proof test.

Obtaining the lightest tank structure possible requires screening the smallest maximum flaw which could exist in a pressure vessel after a proof test. This can be accomplished by either of the following methods: Test the tanks to very near the yield strength at very low temperature (-300 F to -423 F); test the tanks to a stress level in excess of the yield strength of the material. The effect of both methods is shown in Figure A-21. It is readily apparent that testing to a stress level in excess of the yield strength provides for screening of the smaller critical flaw size. However, this also increases the probability of failure during the proof test. Therefore, this analysis will use a low temperature proof (cryoproof) test with a maximum stress equal to 95% of the yield strength at temperature.

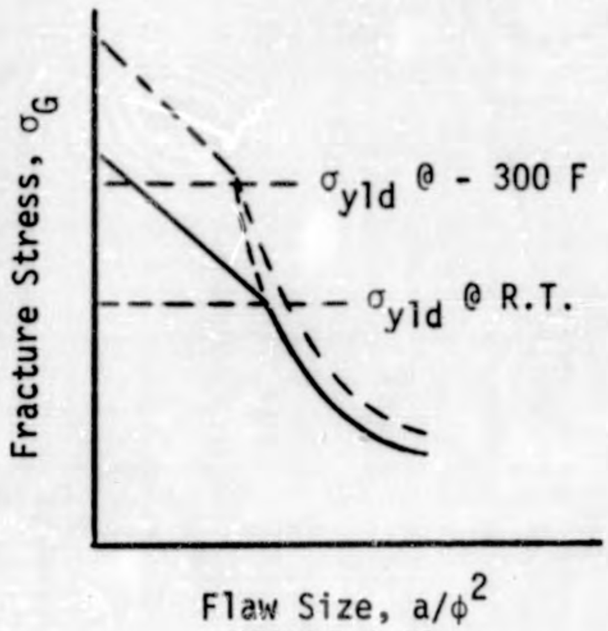


FIGURE A-21

The flaw screened by a cryoproof test of 2014-T651 propellant tanks will be less than 0.02 inches deep. A flaw of this size will be very difficult to find prior to the proof test. As a result, a strong potential exists for failures to occur during proof test. Some of the failures will be flaws which become leakers and other flaws, if sufficiently long, will result in a catastrophic failure of the test specimen. This high risk problem is normally associated with testing of thin material pressure vessels. This problem will be the same regardless of the material, 2014-T651 or 2219-T87, selected for the final TUG design.

With the selection of the proof test stress level (95% F_{ty}) and temperature (T_p < -297), it is possible to determine the flaw sizes screened by the proof test. A strength analysis is used to obtain preliminary sizing of the propellant tank bulkheads and sidewalls. Results of the analysis provided the following thicknesses:

Table A-2. Propellant Tank Skin Thickness

LH_2 Tank	t (in.)	LOX Tank	t (in.)
Bulkheads	0.015	Fwd Bulkhead	0.032
Sidewall	0.034	Aft Bulkhead	0.023

These thicknesses were not among those selected for presentation when developing the Failure Stress and Sustained Load Threshold Stress vs Flaw Size curves (Figures A-15 and A-16). Therefore, it was necessary to perform some interpolation and extrapolation of the data shown in Figures A-15 and A-16 to obtain the critical flaw sizes pertaining to the above thicknesses.

New curves were developed to show critical Flaw Size (a_i and a_{TH}) vs Thickness. The curves were obtained by selecting the critical flaw sizes, under proof/operating stress conditions, from the curves in Figures A-15 and A-16 for the thicknesses 0.025, 0.050 and 0.100 inches. The flaw size/thickness data were plotted to obtain the curves shown in Figures A-22 and A-23. The solid lines shown reflects data interpolation and the dashed lines data extrapolation.

The flaw size screened by a proof test can now be obtained for those thicknesses generated by the strength analysis.

These flaw sizes are summarized below:

Propellant Tank	Flaw Depth a_i (in.)	t (in.)
LH_2 Tank		
Bulkheads	0.012	0.015
Sidewall	0.018	0.034
LOX Tank		
Fwd Bulkhead	0.025	0.032
Aft Bulkhead	0.016	0.023

The next step in the analysis is to obtain the pressure-temperature history anticipated for the TUG vehicle. This provides the information necessary for calculating the flaw growth. Ullage pressure cycles required for the TUG propellant tanks are summarized below. It was determined that the same pressure history could be used for both the LOX and LH_2 tanks.

NOTE: Strength analysis are based on an assumed operating stress.

Having determined the largest permissible flaw screened by a proof test and the pressure-temperature history of the propellant tanks, it is now possible to calculate flaw growth. The analytical procedure used to obtain the total flaw growth is described below.

1. Determine the maximum and minimum stresses associated with each different pressure cycle indicated in Table A-3.

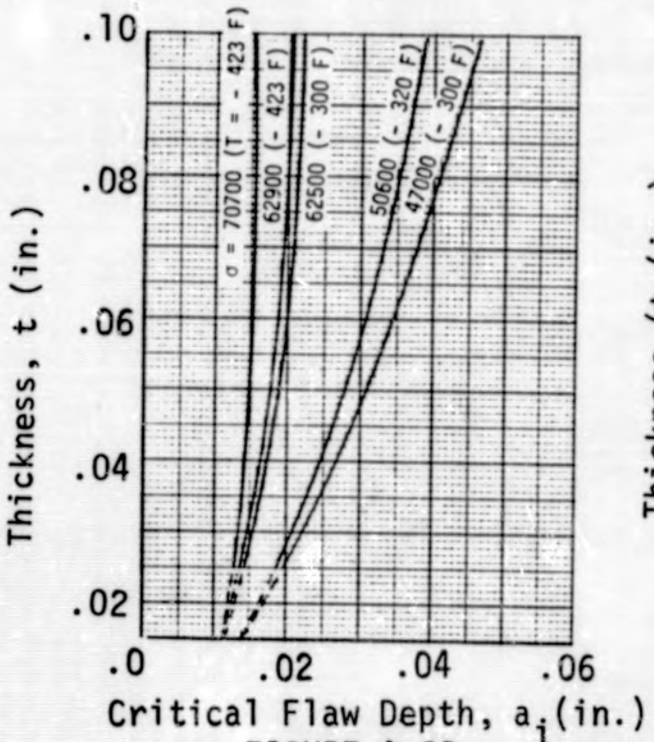


FIGURE A-22

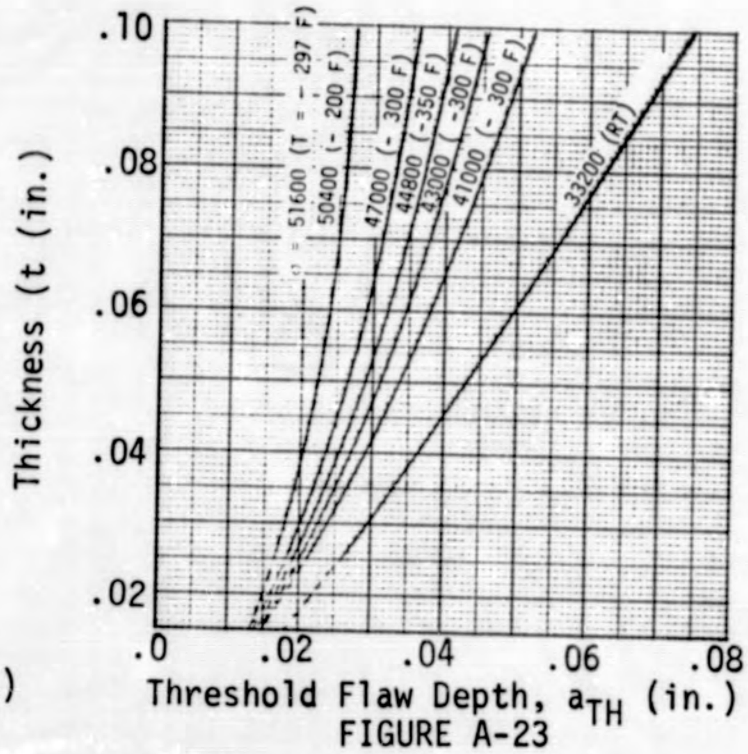
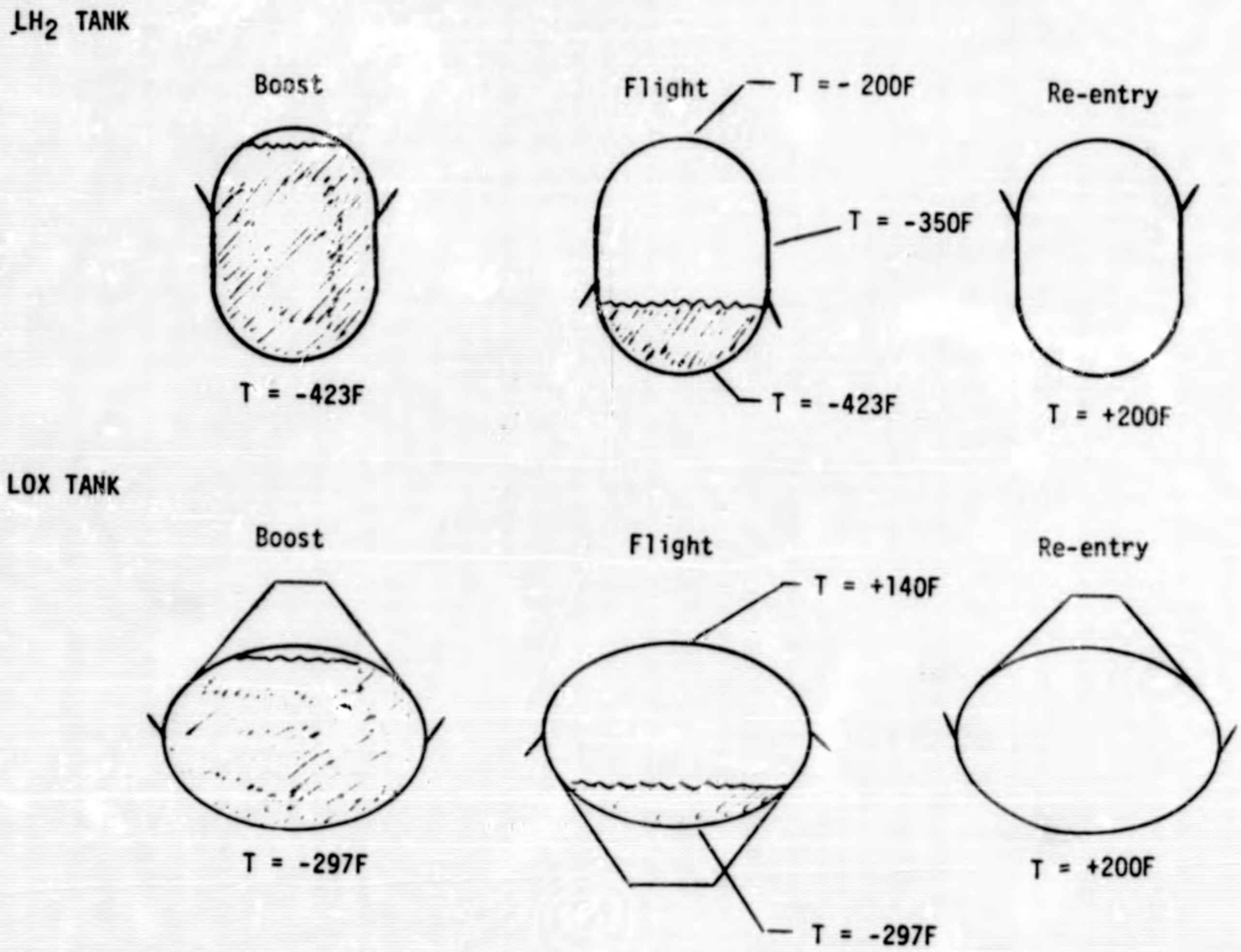


FIGURE A-23



Thermal Data
FIGURE A-24

Table A-3. Propellant Tank Pressure/Temperature Histories

CONDITION	CYCLES	P_{MAX}(PSIG)	P_{MIN}(PSIG)	TEMP °F
STATIC FIRING OPERATION	1	--	--	--
PRETEST CHECKOUT	1	9.3	0	RT
TEST	3	9.3	0	*
POST TEST CHECKOUT	1	9.3	0	RT
FLIGHT MISSIONS	20	--	--	
PREFLIGHT CHECKOUT	1	9.3	0	RT
ORBITER FLIGHT	1	24	0	*
TUG FLIGHT	6	24	15	*
TUG SAFEING	2	17	2	*
POST FLIGHT CHECKOUT	1	9.3	0	RT

*Temperatures for these conditions vary from the top to the bottom of the tank and are shown in Figure A-24.

2. Determine ΔK_i where

$$\Delta K_i = [1 + .12\ (1 - \frac{a_i}{c}]\ \Delta\sigma_i \sqrt{\frac{\pi\ a_i}{Q}}\ M_{K_i}$$

a_i = Flaw depth = $a_{initial} + \Sigma\Delta\ a_i$

c = One-half the flaw length

$\Delta\sigma_i$ = Cyclic stress range = $\sigma_{max} - \sigma_{min}$

$$Q = \phi^2 - .212 \left(\frac{\sigma_{max}}{\sigma_y}\right)^2$$

$$\phi^2 = \text{Flaw normalizer} = \int_o^{\pi/2} \left[1 - \frac{c^2 - a^2}{ac^2} \sin^2\theta\right]^{1/2} d\theta$$

σ_y = Yield strength of the material

M_{k_i} = Kobayashi far surface correction factor

i = Pressure cycle number

NOTE: The prestatic firing pressure test will provide the pressures needed to calculate $\Delta\sigma_i$. The size of the flaw screened by the proof test will be used for a_i and is to determine M_{k_i}. Subsequent values of $\Delta\sigma_i$, a_i, and M_{k_i} depend on the pressure associated with the new cycle number and the ever larger flaw size.

3. Determine the flaw growth, Δa_i

where

$$\frac{\Delta a_i}{\Delta N} = \frac{C\ (\Delta\ K_i)^n}{(1 - R)\ K_{IC} - \Delta\ K_i}$$

ΔN = 1 (See note below)

K_{IC} = Stress Intensity Range

R = Stress Ratio $(\sigma_{min}/\sigma_{max})$

C,n = Empirical Coefficients (See Table 4)

Table A-4. Empirical Coefficients

Temp	K_{IC}	C	n
RT	40,000	7.68×10^{-15}	3.41
-320°F	44,000	3.00×10^{-13}	2.95

NOTE: The value for the number of cycles (ΔN) remains one for all growth calculations. If a ΔN larger than one were used, it would be unconservatively assumed that the flaw growth rate was constant with increasing flaw size. This is unconservative because the flaw growth rate increases with increasing flaw size.

4. Determine the final flaw size and compare with the tank skin thickness and the sustained load threshold critical flaw size (Figure A-13) for the selected operating stress. If the final flaw size does not exceed either of the above criteria, the skin thickness is accepted. If the flaw exceeds either of the above criteria, the skin thickness

is arbitrarily increased 10%. This serves two purposes; increases the skin thickness; and decreases the operating stress. Steps 2 through 4 are then repeated until a satisfactory skin thickness is obtained.

Results of the analysis are summarized in Table A-5 where the component skin thickness is compared with: the flaw size screened by the proof test; the flaw growth; the total flaw size at the end of 20 missions; and the sustained load threshold critical flaw size at the maximim operating stress.

Table A-5. Flaw Growth Summary

Component	t_{min} (in.)	$a_{initial}$ (in.)	Δa (in.)	a_{TOT} (in.)	a_{TH} (in.)	σ_{oper} (psi)
LH_2 Tank						
Bulkheads	.020	.012	.0018	.014	.015	53,600
Sidewall	.045	.018	.0012	.020	.027	44,800
LOX Tank						
Fwd Bulkhead	.040	.025	.0010	.026	.029	41,000
Aft Bulkhead	.030	.016	.0016	.018	.023	43,000

It should be noted that there is a significant difference between the total flaw size a_{TOT} and the critical threshold flaw size a_{TH}. This margin would be minimized in final design when the fracture mechanics properties of the selected material have been more accurately depicted.

VOLUME III

PART 2

APPENDIX B

FAILURE MODE EFFECT ANALYSIS

TABLE B-1. **FAILURE MODE EFFECT ANALYSIS**

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Vehicle Status	Computer: No output, erroneous output	Loss of vehicle status knowledge. Erroneous assumption of vehicle availability and/or lack of failure reporting. Loss of tug and payload objectives.	IIA
	Memory Unit: Loss of information	Inability to perform on board checkout (OBCO) and make go-no go decisions. Loss of tug and payload objectives.	IIA
	Data Acquisition Units: No output, erroneous output	Loss of one subsystem status or error in one subsystem status. Effects are dependent upon which subsystem information is lost and at what mission phase loss occurs.	IIB
	Measurement Processing Units: No output, erroneous output	Loss of ability for uplink command and downlink transmission.	III
	Status Sensor Transducers: No output, erroneous output	Loss of measurement information or erroneous measurement information. Critical measurements will have two or more sensors.	III
	Signal Conditioners: No output, erroneous output	Loss of measurement information or erroneous measurement information. Critical measurements will have two or more sensors.	III
	Power Components	Loss of all vehicle status to loss of one measurement.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Sequence and Control Management	Computer: No output	Loss of ability to control vehicle or initiation of wrong control signals, loss or confusion of programmed sequencing.	IIA
	Computer: Output Error	Command sequences that may result in a hazard to the orbiter (engine fire, etc.) will require <u>both</u> computer and uplink commands. Loss of Tug & Paylod objectives.	
	Memory Unit: Loss of information	Loss of sequence and control steps. Loss of tug and payload objectives.	IIA
	Data Acquisition Units: No Output	Loss of ability to transmit sequence and control signals, loss of tug and payload objectives.	IIA
	Data Acquisition Units: Output Error	Commands are sent and verified by computer before execute	
	Measurement Processing Units: No output	Loss of ability to update tug control sequences. Loss of tug and payload objectives.	IIA
	Status Sensor Transducer: No Output	Loss of status or erroneous status information for one parameter. Critical parameters will have two or more sensors.	III
	PM Transponder: No Output	Loss of uplink command and data; erroneous uplink commands and data. Logic coding will assure acceptance of only viable commands.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Sequence and Control Management (Continued)	Decoder: No output	Loss of uplink command and data; erroneous uplink commands and data. Logic coding will assure acceptance of only viable commands.	IIA
	Omni Antenna: No output	Loss of uplink command and data; erroneous uplink commands and data. Logic coding will assure acceptance of only viable commands.	IIA
	Hybrid Junction: No output	Loss of uplink command and data.	IIA
	Isolation Filter: No output, short	Loss of uplink command and data, addition of noise and distortion to the system.	IIA
	Status Sensor Transducers: No output	Failure to send required command due to sensor failure. Initiation of wrong command due to erroneous sensor output. Critical parameters will have redundant sensors	III
	Power Components	Loss of tug and payload objectives.	IIA
On-Board Checkout (OBCO)	Computer: No output, erroneous output.	Loss of checkout ability or wrong go-no go indication.	III
	Memory Unit: Loss of Information	Loss of checkout ability or wrong go-no go indication.	III
	Data Acquisition Unit: No output.	Loss of OBCO for one subsystem, mission abort required.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
On-Board Checkout (OBCO)	Measurement Processing Unit: No output, erroneous output	Loss of ability to transmit uplink and downlink data on transmission of erroneous data. Would require mission abort.	IIB
	Interface Unit: No output	Loss or erroneous checkout of one subsystem. Mission abort.	IIB
	Status Sensor Transducer: No Output	Loss or error in one parameter. Critical parameters will have redundant sensors.	III
	Signal Conditioners: No output	Loss or error in one parameter. Critical parameters will have redundant sensors.	III
	Power Components	Mission abort, loss of mission objectives.	IIB
Redundancy Mgt.	Computer: No Output Output Error	Loss of ability to switch to a redundant unit or switching when not required. Potential loss of tug mission objectives.	IIB
	Memory Unit: No Output	Loss of ability to discriminate between working and non-working redundant units.	IIB
	Data Acquisition Unit: No output Output Error	Loss of ability to switch to a redundant unit or switching when not required. Potential loss of tug mission objectives.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Redundancy Mgmt.	Interface Unit: No output	Loss of ability to switch to a redundant unit or switching when not required. Potential loss of tug mission objectives.	IIB
	Measurement Processing Unit: No output.	Loss of backup redundancy control or an inadvertent switch to a redundant unit.	III
	Status Sensor Transducers: No output	Loss of ability to switch to a redundant unit when required or inadvertent switching. Critical parameters will have redundant sensors.	III
	Signal Conditioners: No output	Loss of ability to switch to a redundant unit when required or inadvertent switching. Critical parameters will have redundant sensors.	III
	Power Components:	Mission abort, loss of mission objectives.	IIB
Propellant Mgmt. and Utilization	Computer: No Output, Erroneous Output	Loss of ability to maintain mixture ratios; erroneous mixture ratios; possible mission loss.	IIB
	Memory Unit: No Output	Loss of ability to maintain mixture ratios; erroneous mixture ratios; possible mission loss.	IIB
	Data Acquisition Unit: No output-Output error	Loss of ability to maintain mixture ratios; erroneous mixture ratios; possible mission loss.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Mgmt. and Utilization	Measurement Processing Unit: Output loss/error	Loss of ability to monitor propellant tank measurements and process data to prescribed schedules. Possible mission loss.	IIB
	Interface Unit: Output loss/error	Loss of ability to maintain main engine mixture ratios. Erroneous mixture ratios,information. Possible tug mission loss.	IIB
	Status Sensor Transducers: Output loss/error	Degraded engine mixture ratio control for the main engines.	III
	Signal Cond. Unit Output loss/error	Degraded engine mixture ratio control for the main engines.	III
	Power Components No Output	Mission abort, loss of mission objectives.	IIB
Propellant Conditioning and Transfer	Computer: Output loss/error	Loss of delta V or APS, loss of tug mission objectives.	IIB
	Memory Unit: Output loss/error	Loss of delta V or APS, loss of tug mission objectives.	IIB
	Data Acquisition Output loss/error	Loss of delta V or APS , loss of tug mission objectives.	IIB
	Interface Unit Output loss/error	Loss of delta V or APS , loss of tug mission objectives.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Conditioning and Transfer	Measurement Processing Unit: No Output	Loss of ability to monitor system measurements and maintain nominal system operation. Loss of delta V or APS, loss of tug mission objectives.	IIB
	Status Sensor Transducers: No Output	Loss of conditioning requirement information or erroneous information. Loss of APS, loss of tug mission objectives.	IIB
	Signal Conditioner: No Output	Loss of conditioning requirement information or erroneous information. Loss of APS, loss of tug mission objectives.	IIB
	Power Components No Output	Mission abort, loss of mission objectives.	IIB
Propellant Orientation	Computer: Output loss/error	Loss of thrust for propellant setting, loss of ability to start main engines. Loss of mission objectives.	IIB
	Memory Unit No output	Loss of thrust for propellant setting, loss of ability to start main engines. Loss of mission objectives.	IIB
	ACPS Driver Electronics: No Output Output Error	Loss of thrust for propellant setting, loss of ability to start main engines. Loss of mission objectives. APS thrusters are redundant, therefore, the driver electronics failure must be such that all electronics are negated, unlikely.	III
	Power Components No Output	Loss of APS burn, loss of mission objectives.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Safing	Computer: Output loss/error	Tug will not be safed for re-entry with orbiter. Unsafe condition will exist. Loss of mission objectives if safing is initiated prematurely.	IIB
	Memory Unit: No Output	Tug will not be safed for re-entry with orbiter. Unsafe condition will exist. Loss of mission objectives if safing is initiated prematurely.	IIB
	Data Acquisition Unit: Output loss-Output error	Tug will not be safed for re-entry with orbiter. Unsafe condition will exist. Loss of mission objectives if safing is initiated prematurely.	IIB
	Interface Units Output loss/error	Tug will not be safed for re-entry with orbiter. Unsafe condition will exist. Loss of mission objectives if safing is initiated prematurely.	IIB
	Measurement Processing Unit: Output loss/error	Loss of ability to safe vehicle from external command. Loss of tug mission objectives.	IIB
	Status Sensor Transducers: Output loss/error	Loss of verification of safing or wrong safing indication. Critical parameters will have redundant transducers.	III
	Signal Cond. Unit: Output loss/error	Loss of verification of safing or wrong safing indication.	III
	PM Transponder: Output loss/error	Loss of external safing command, loss of safing verification. Unsafe condition. FM system transmits safe verification as backup and vehicle has self safe initiation program.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Safing (Continued)	Decoder Recorder: No Output Output Error	Loss of external safing command, loss of safing verification. Unsafe condition. FM system transmits safe verification as backup and vehicle has self safe initiation program.	III
	Hybrid Junctions Open, Short	Loss of external safing command, loss of safing verification. Unsafe condition. FM system transmits safe verification as backup and vehicle has self safe initiation program.	III
	Isolation Filter Open, Short	Loss of external safing command, loss of safing verification. Unsafe condition. FM system transmits safe verification as backup and vehicle has self safe initiation program.	III
	Power Components No Output	Loss of safing ability	IIB
Navigation	Computer: Processing Error	Loss of orbit and position determination. Loss of mission and payload objectives.	IIA
	Memory Unit: No Output	Loss of orbit and position determination. Loss of mission and payload objectives.	IIA
	Data Acquisition Unit: No Output Output Error	Loss of mission and payload objectives.	IIA
	Interface Unit: No Output Output Error	Loss of mission and payload objectives.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Navigation (Continued)	Measurement Processing Unit: Output loss/error	Loss of external updating ability, position and orbit degradation, erroneous update.	IIB
	Inertial Measurement Unit: Output loss/error	Loss of position or erroneous position information. Time, orbit, and position error. Loss of Tug and Payload objectives.	IIA
	Star Tracker: No Output Output Error	Loss of fine position update, erroneous position update. Attitude and State vector error. Loss of Tug and Payload objectives.	IIA
	Horizon Sensor: Output loss/error	Loss of orbit radius update or erroneous update. State Vector errors. Loss of Tug and Payload objectives.	IIA
	Autocollimater: Output loss/error	Loss of vehicle distortion data. Error in IMU update. State Vector error. Loss of Tug & Payload objectives.	IIA
	PM Transponder: Output loss/error	Loss of navigation backup. State vector error.	III
	Decoder: Output loss/error	Loss of navigation backup. State vector error.	III
	Omni Antenna: No Output	Loss of navigation backup. State vector error.	III
	Isolation Filter: No Output	Loss of navigation backup. State vector error.	III
	Hybrid Junction No Output	Loss of navigation and backup. State vector error.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Navigation (Continued)	FM Transmitter: Output loss/error	Loss of downlink data, distortion or noise in downlink.	III
	Power Amplifier: Output loss/error	Loss of downlink data, distortion or noise in downlink.	III
	R.F. Multiplexer: Output loss/error	Loss of downlink data, distortion or noise in downlink.	III
	Power Components: Output loss/error	Loss of navigation and position capability, loss of Tug and Payload objectives.	IIA
Guidance	Computer: Output loss/error	Loss of or wrong control outputs. Loss of Tug and Payload objectives.	IIA
	Memory Unit: No Output	Loss of Guidance Control signal. Loss of Tug and Payload objectives.	IIA
	Data Acquis. Unit: Output loss/error	Loss of Guidance Control Signal. Loss of Tug and Payload objectives.	IIA
	Interface Unit: Output loss/error	Loss of Guidance Control Signal. Loss of Tug and Payload objectives.	IIA
	Power Components No Output	Loss of orbit change ability. Stabilization will be provided by battery backup.	IIB

TABLE B-1. **FAILURE MODE EFFECT ANALYSIS**

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Rendezvous	Computer: Output loss/error	Loss of ability to enter rendezvous orbit with payload to be retrieved. Loss of tug objectives.	IIB
	Memory Unit: No Output	Loss of ability to enter rendezvous orbit with payload to be retrieved. Loss of tug objectives.	IIB
	Data Acquis. Unit Output loss/error	Loss of ability to enter rendezvous orbit with payload to be retrieved. Loss of tug objectives.	IIB
	Interface Unit: Output loss/error	Loss of ability to enter rendezvous orbit with payload to be retrieved. Loss of tug objectives.	IIB
	ACPS Driver Electronics: Output loss/error	Loss of orientation and stabilization ability. Loss of tug and payload objectives.	IIA
	Lasar Radar: No Output	Loss of ability to perform rendezvous maneuver. Loss of tug objectives.	IIB
	PM Transponder Output loss/error	Loss of uplink communication during rendezvous. Loss of vehicle objectives. (Backup system)	III
	Decoder: Output loss/error	Loss of uplink communication during rendezvous. Loss of vehicle objectives. (Backup system)	III
	Omni Antenna: No output	Loss of uplink communication during rendezvous. Loss of vehicle objectives. (Backup system)	III
	Isolation Filter: No Output	Loss of uplink communication during rendezvous. Loss of vehicle objectives. (Backup system)	III

TABLE B-1. **FAILURE MODE EFFECT ANALYSIS**

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Rendezvous (Continued)	Hybrid Junction No Output	Loss of uplink communication during rendezvous. Loss of vehicle objectives. (backup system)	III
	Power Amplifier No Output	Loss of downlink communications during rendezvous. Loss of vehicle objectives. (back up system)	III
	F.M. Transmitter Output loss/error	Loss of downlink communications during rendezvous. Loss of vehicle objectives. (back up system)	III
	R.F. Multiplexer Output loss/error	Loss of downlink communications during rendezvous. Loss of vehicle objectives. (backup system)	III
	R.F. Switch: No Output, Unable to Switch	Loss of video signal, loss of docking ability, loss of tug objectives. (Backup system)	III
	Bi-Phase Modulator: No Output	Loss of downlink data.	III
	EOS Rendezvous Transponder: No Output	Loss of ability for range and range rate closure of tug. Degraded shuttle - tug rendezvous. Pilot visual backup.	III
	EOS Rendezvous Antenna: No Output	Loss of ability for range and range rate closure of tug. Degraded shuttle - tug rendezvous. Pilot visual backup.	
	Power Components: No output	Loss of rendezvous, loss of tug objectives.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Docking	Computer: No Output Erroneous Output	Loss of docking monitor ability, primary control command is by "man-in-loop" limited by computer mode monitor.	III
	Memory Unit: No Output	Loss of docking monitor ability, primary control command is by "man-in-loop" limited by computer mode monitor.	III
	Data Acquisition Unit: No Output Erroneous Output	Loss of docking monitor ability,primary control command is by "man-in-loop" limited by computer mode monitor.	III
	Interface Unit: No Output, Erroneous Output	Loss of docking monitor ability, primary control command is by "man-in-loop" limited by computer mode monitor.	III
	Measurement Processing Unit: Output loss/error	Loss of docking monitor ability, primary control command is by "man-in-loop" limited by computer mode monitor.	III
	ACPS Driver Electronics: No Output, Erroneous Output	Fourteen driver circuits, inherent control capability exists for loss of any one driver circuit.	III
	Laser Radar: No Output	Loss of ability to dock. Loss of tug objectives. Television is backup.	III
	Television Camera: No Output, Erroneous Output	Loss of ability to dock. Loss of tug objective.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Docking (Continued)	Docking Sensors No Output, False Output	Loss of ability to assure good dock or indication of dock when not secure. Loss of tug & payload. Redundant sensor plus procedural check.	III
	FM Transponder No Output Erroneous Output	Loss of uplink command, possible error command, loss of tug mission objectives. Failure will initiate abort mode, closure rates are too low to cause extensive damage. Loss of tug objectives.	IIB
	Decoder: No Output, Erroneous Output	Loss of uplink command, possible error command, loss of tug mission objectives. Failure will initiate abort mode, closure rates are too low to cause extensive damage. Loss of tug objectives.	IIB
	Omni Antenna: No Output	Loss of uplink command, possible error command, loss of tug mission objectives. Failure will initiate abort mode, closure rates are too low to cause extensive damage. Loss of tug objectives.	IIB
	Power Amplifier No Output Erroneous Output	Loss of downlink data, loss of video, loss of tug objectives.	IIB
	FM Transmitter No Output, Erroneous Output	Loss of downlink data, loss of video, loss of tug objectives.	IIB
	RF Switch: No Output	Loss of downlink data, loss of video, loss of tug objectives.	IIB
	Isolation Filter: No Output	Loss of uplink command data. Loss of tug objectives.	IIB.

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Docking (Continued)	Hybrid Junction No Output	Loss of uplink command data. Loss of tug objectives.	IIB
	Bi Phase Modulator: No Output	Loss of down link data. Loss of tug objectives.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Docking (Continued)	Power Components: No Output	Loss of docking capability, loss of tug objectives.	IIB
Thermal Control (ECS)	Thermostat: Short, Open	Heaters continuously on, heat sink raises Avionics temperature above maximum allowable. Possible Avionics failure. Heaters continuously off, heat sink drops below minimum; Avionics failure. Thermostats are redundant.	III
	Heater: On, Off	Heaters continuously on, heat sink raises Avionics temperature above maximum allowable. Possible Avionics failure. Heaters continuously off, heat sink drops below minimum - Avionics failure. Heaters are redundant.	III
	Louvers: Open, Closed	Loss of one louver is approximately 8% of total louver ability. Louvers are approximately 50% of thermal control ability. Mechanical (Bi-Metalic) system.	III
Power Generation and Control	Fuel Cell		
	1. Loss of Output	Loss of primary power to using functions - emergency battery will supply energy requirements for one-half hour.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Power Generation and Control (Continued)	Battery		
	1. Loss of Output	Loss of backup power capability (primary power supply is unaffected due to blocking diodes)	III
	2. Cell Fracture or Rupture	Possible damage to adjacent assemblies or components through the release of electrolyte or pressure propelled fragments. Battery is in an insulated case and is equipped with a pressure relief valve.	III
	3. Loss of Monitor Temp.	Loss of information only	III
Power Distribution	Power Transfer Switch		
	1. Fails Open or Inadvertently opens	Loss of all power to using functions - (both the fuel cell and battery power circuits would become disabled)	IIA
	2. Fails Closed or Inadvertently closes.	Emergency backup system distributor Avionics will be powered prematurely by the emergency battery. Power reduction in battery life.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Power Distribution (Continued)	Power Control Switch (PCS) Approx. 95 Units		
	1. Fails Open or Inadvertently Opens	Loss of subsystem functions receiving electrical power through the specific PCS. NOTE: Refer to specific system function for individual criticality categories.	IIA
	2. Fails Closed or Inadvertently Closes	Premature powering of a normal subsystem function. Inability to safe (de-energize) the subsystem function	IIB
Attitude Stabilization and Control	Central Processor Unit:		
	Loss of Output or Erroneous Output.	*Loss of autonomus stabilization capability. No output to attitude control. Possible inducement of unwanted pitch, yaw or roll to tug/payload.	IIA
	Memory Unit:		
	Loss of output or erroneous output	*Loss of attitude reference. Restricts payload retrieval and docking. Possible inducement of unwanted pitch, yaw or roll to tug/payload.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Attitude Stabilization and Control (Continued)	Data Acquisition Unit: Loss of Output or Erroneous Output	*Loss of autonomous stabilization capability. No attitude control or data feedback. Possible inducement of unwanted pitch, yaw or roll to tug/payload.	IIA
	Interface Unit Loss of Output	*Loss of autonomous stabilization capability. No attitude control or data feedback.	IIA
	Data Bus Loss of Output	*Data and commands cannot be transmitted.	IIA
	Power Components Loss of Output	Loss of attitude stabilization and control function. Loss of payload objectives.	IIA
	* BACKUP RATE STABILIZATION (ATTITUDE HOLD) IS AVAILABLE FOR VEHICLE STABILIZATION IN THE EVENT OF A PRIMARY SUB-ASSEMBLY OR COMPONENT FAILURE.		
	Internal Meas. Unit: Loss of Output or Erroneous Output	Loss of attitude reference. Backup rate gyro will take over. Possible unwanted pitch, yaw or roll thrust.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Attitude Stabilization and Control	Star Tracker : Loss of Output or Erroneous Output	Degrades attitude reference. Loss of ability to update for IMU attitude stabilization drift. Could induce an improper attitude.	IIB
	Horizon Sensor: Loss of Output or Erroneous Output	Loss of ability to correct IMU to reference position (for navigation) uses ground track and uplink as alternate. Could induce an improper attitude.	III
	Backup Rate Gyro: Loss of Output or Erroneous Output	Inability to provide stabilization data following a primary system failure.	III
	Backup Rate Stabilization Logic: Loss of Output or Erroneous Output	Inability to translate backup rate gyro data into stabilization outputs following a primary system failure.	III
	APS Driver Electronics:		
	1. Loss of Output	Loss of ability to command the attitude jets. (Loss of single driver would not cause mission loss)	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Thrust Vector Control	Central Processing Unit: Loss of Output or Erroneous	Loss of ability to compute updated control sequence during main engine burn. Erroneous control data could result in a course deviation.	IIA
	Memory Unit: Loss of Output or Erroneous Output	Loss of reference course data and operational programs. Erroneous control data could result in a course deviation.	IIA
	Data Acquisition Unit: Loss of Output or Erroneous Output	Loss of command and measurement data transfer and retention capability. Erroneous control data - could result in a course deviation.	IIA
	Interface Unit: Loss of Output	Loss of command and measurement data conversion capability at the DAU/Data Bus Interface	IIA
	Data Bus: Loss of Output	Inability to transmit data and commands for main engine.	IIA
	Power Components: Loss of Output	Loss of power to components/assemblies providing the functional capability.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG AVIONICS

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Thrust Vector Control (Continued)	Inertial Measuring Unit: Loss of Output or Erroneous Output	* Inability to accurately maintain course through main engine maneuvering. Induces an inaccurate course maneuver.	IIA
	Backup Rate Gyro: Loss of Output or Erroneous Output	Inability to prevent or arrest tumbling following primary system failure.	III
	TVC Driver Electronics: Loss of Output or Erroneous Output	Loss of capability to command main engine gimballing.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Supply	APS Feedout Tank (LH_2 & LOX)		
	1. Leakage	Loss of propellant supply would result in degraded operation. Time limited attitude hold (30 min.) is available from gaseous propellant accumulators.	IIA
	APS Feedout Tank Refill/Vent Control System (LH_2 & LOX)		
	1. Fails to fill tank	Loss of propellant supply would result in degraded operation. Time limited attitude control (30 min.) available from gaseous propellant accumulators	IIA
	GH_2/GOX Accumulators (1250 PSI)		
	1. Rupture/ Explosion	Loss of vehicle attitude control. Possible damage to vehicle payload or orbiter.	IA
	2. External Leakage	Degraded operation-vehicle attitude control could be lost during latter phases of the mission	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Feed	Tank Isolation Control Valves N.O.(LH_2 & LOX)		
	1. Fails to remain in open position.	Loss of main propellant supply - results in degraded operation. Vehicle attitude control and fuel cell would be lost when accumulator supply becomes depleted. Valve not operated during flight. (Ground power operated only)	IIA
	Turbopumps (LH_2 & LOX)		
	1. Loss of output	Loss of vehicle attitude control and fuel cell would result when accumulator supply becomes depleted.	IIA
	Turbopump By Pass Bleed Valve (LH_2 & LOX)		
	1. Fails to open	No bypass flow would result in turbopump cavitation at start - pump stall and flow cessation. Loss of vehicle attitude control and fuel cell would result when system accumulator supply becomes depleted.	IIA
	2. Fails to close	Insufficient propellant supply to heat exchanger resulting in possible overheating of heat exchanger. System accumulators could not be fully maintained resulting in degraded operation of the system.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Feed (Continued)	Turbopump Gas Generators (GG) (LH_2 & LOX Systems)		
	1. Propellant Control Valves Fail to Open	GG fails to provide hot gas for turbo-pump operation. System accumulators could not be fully maintained resulting in degraded operation of the system.	IIA
	2. Propellant Control Valves Fail to Close	Inability to shutdown turbine drive GG could lead to overpressurization of accumulator. Vehicle propellant isolation valves operate simultaneously with valves providing closure redundancy.	III
	3. Ignition Failure	Failure of GG to ignite and maintain combustion would result in insufficient propellant supply to the accumulators. System operation would be degraded. Loss of vehicle attitude control and fuel cell would result when system accumulator supply becomes depleted. (30 min. attitude hold) Redundant ignition and spark plugs will be considered to preclude occurrence of failure mode.	IIA
LOX Turbo Pump He Purge	He Receiver (Engine Support System) (3000 PSI) Rupture Explosion	Loss of pump purge. Pump and vehicle damage could result. NOTE: He tank used for main engine purge and APS turbo-pump purge	IA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
LOX Turbo Pump He Purge	Control Valve		
	1. Fails closed	Purge of pump intermediate seals separating the pump/turbine and actuation of pump lift-off seals would be terminated. Pump damage could result. Time limited attitude hold (30 min.) is available from gaseous propellant accumulators.	IIA
	2. Fails open	Purge system He capacity would be depleted. Latter phases of the mission could experience conditions noted above.	IIA
Propellant Feed	Series Check Valves (LH_2 & LOX)		
	1. Fails to open	Loss of main propellant supply to system accumulators. Loss of vehicle attitude control and fuel cell would result when system accumulator supply becomes depleted.	IIA
	2. Fails to close	Back flow of accumulator pressurant through propellant tank by pass valve and pump. Excessive pressure at pump inlet & duct may cause structural failure. Check valves are series redundant to preclude this failure condition from occurring.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Feed (Continued)	Feedline Thermal Control Valves		
	1. Fails to allow circulation of cooling (LH_2)	Conditioning of LH_2 or LOX feedlines would be terminated resulting in degraded operation of the turbo-pump feed system. System operation capability would be reduced.	IIB
	Heat Exchanger (LH_2 & LOX) 1. Rupture	Heat exchanger element burn through as a result of heat exchanger GG failure to cutoff, turbine GG failure to ignite, or coolant flow failure. Loss of propellant would result in degraded operation. Time limited attitude control (30 min hold) available from gaseous propellant accumulators.	IIA
	Heat Exchanger Gas Generator(GG) (LH_2 & LOX)		
	1. Propellant control valves fail to open	1. Failure to provide hot gas for heat exchanger operation APS and fuel cell would become inoperative when accumulator supply becomes depleted.	IIA
	2. Propellant Control Valves fail to close	2. Inability to shutdown heat exchanger could possibly lead to overheating of the exchanger. Vehicle propellant isolation valves operate simultaneously with G.G. valves providing closure redundancy.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Feed (Continued)	Burst Disk Relief Valve (GOX & GH_2)		
	1. Rupture	Rupture of the high pressure relief disk would cause the overboard bleed of gaseous propellant. Vehicle attitude control and fuel cell operation could be affected during latter phases of the mission.	IIB
	Safety Pressure Switch (GOX & GH_2) High System Pressure		
	1. Fails in Actuated Position	Accumulator propellant would be bled overboard and GG operation would be terminated. If failure occurs during orbiter boost period, turbo-pump and heat exchanger GG operations could be affected resulting in the failure to replenish accumulators for normal APS and fuel cell operation.	IIA
	GG Pressure Switch Propellant Control		
	1. Fails in Actuated Position	Turbo-pump and heat exchanger GG operation would remain shutdown. Time limited attitude hold (30 min.) is available from gaseous propellant accumulator.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Feed (Continued)	2. Fails to actuate upon increasing system pressure (open)	System pressure would increase causing safety pressure switch control system to become activated. No direct effect upon system operation.	III
	Pressure Regulator Assembly (GOX and GH_2) (375 PSI Outlet Pressure)		
	1. Low Outlet Pressure	Inadequate pressurization would result in sluggish performance of the attitude control system. Loss of outlet pressure would disable the attitude control and fuel cell systems. Parallel redundant regulators are utilized to reduce probability of this failure mode occurring.	III
	2. High Outlet Pressure	System pressure would activate overboard bleed and terminate GG operation. APS and fuel cell operation would be terminated. Series redundant regulators are utilized to reduce probability of this failure mode occurring.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Feed (Continued)	Pressure Switch (GOX and GH_2) Low System Pressure		
	1. Fails in Actuated Position	Accumulator propellant would be bled overboard and fuel cell operation propellant would be terminated. Vehicle attitude control could be affected during latter phases of the mission.	IIB
	2. Fails to Actuate upon Increasing System Pressure(Open)	Increasing system operating pressure (GOX or GH_2) could cause uneven thruster combustion resulting in possible damage to thruster. Vehicle structural damage could result. Second order failure - pressure regulator must fail to maintain adequate pressure level.	III
Controlled Thrust	APS Engine Thruster Assemblies (14)		
	1. Propellant Control Valves Fail to Open or Close	Failure of an engine valve (GH_2 or GOX) to open or failure of the series isolation valves to close would cause loss of affected thruster. Vehicle attitude control will be maintained by remaining thrusters.Series isolation valves provide closure redundancy.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: AUXILIARY PROPULSION SYSTEM (APS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Controlled Thrust (Continued)	2. Propellant Control Valves Fail to Remain Closed. (Excessive Leakage)	Loss of gaseous propellant would contribute towards unwanted thrust and accelerated depletion of the vehicle propellant supply resulting in eventual loss of the vehicle attitude control. Series valves provide closure redundancy.	III
	3. Engine Burn Through	Damaged engine would allow hot gas to escape in uncontrolled condition resulting in possible vehicle structural damage.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: PROPELLANT TANK (LH_2 & LOX) PRESSURIZATION SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Tank Pressurization	Prepressurization Solenoid Valves (GH_2 & GOX)		
	1. Fail Open or Leak	No mission effect. Series redundant valves prevent loss of APS accumulator pressurant when a valve fails to close.	III
	2. Fail Closed	Loss of ability to prepressurize main LH_2/LOX tanks for engine start. Loss of tug mission objectives.	IIB
	Pressure Regulators (Two Stage) (GH_2 & GOX)		
	1. High Outlet Pressure	Propellant tank pressure relief valves would control tank pressure to a safe level during engine operation. Pressurizing gas (propellant) would be lost overboard through the vent which could cause loss of mission objectives.	IIB
	2. Low Outlet Pressure	Degraded engine performance or loss of main engine operation. Loss of tug mission objectives.	IIB
	Engine Isolation Check Valves (GH_2 & GOX)		
	1. Fail or Remain Open During Operation	Degraded engine performance or loss of main engine operation.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPELLANT TANK SAFING & VENTING

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Main Propellant Tank Venting	Selector Valves for Venting Tanks from Top or Bottom (LH_2 & LOX Tanks)		
	1. Fail Closed	No mission effect - valves are parallel redundant.	III
	2. Fail Open	Possibility of some propellant loss during vent mode.	IIB
	Propellant Tank Pressure Relief Vent Valves (Normal (low) mode 17 PSIA) (Main Engine Operation (High) Mode 24 psia)		
	1. Fail Open	Loss of propellant tank pressurization would result in failure of main engine operation	IIA
	2. Fail Closed	Parallel redundancy - No effect.	III
	Vent Valve (High & Low Mode) Solenoid Valves (LH_2 & LOX Tank)		
	1. Fail to Operate	High pressure mode of vent valve could not be obtained resulting in failure to start main engines	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPELLANT TANK SAFING & VENTING

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Tank Purge & Blowdown for Safing	Motor Driven (Open/Closed) Shutoff Vent Valve (LH_2 & LOX Tank)		
	1. Fails Closed	Fill and drain system may be used as back-up for propellant tank blow down with operational constraint	III
	2. Fails Open	Inability to pressurize tank for re-entry resulting in tank collapse on re-entry.	IIB
	Fill/Drain Valves (LH_2 & LOX)		
	1. Fails to operate open	Parallel redundancy is provided. No effect upon filling or draining.	III
	2. Fails to close	Propellant tanks cannot be pressurized for engine operation or for purge operations.	IIB
	Non-Propulsive Vent Control Valves(LH_2 & LOX Tank)		
	1. Fail Open	Venting of propellant tanks during blowdown would allow propellant vapors to enter the orbiter cargo bay which can result in a hazardous condition to the orbiter crew.	IIB.

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPELLANT TANK SAFING & VENTING

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Tank Purge & Blowdown for Safing (Continued)	2. Fail Closed	No mission effect - parallel redundancy	III
	He Solenoid Control Valve (Pressure Switch Controlled) (LH_2 & LOX Tanks)		
	1. Fails Closed	Inability to purge propellant tanks	IIB
	2. Fails to Remain Closed	Loss of He purge supply - inability to purge propellant tanks.	IIB
	He Solenoid Valve Pressure Switch (LH_2 & LOX Tanks)		
	1. Fails to Actuate	Unable to pressurize and purge propellant tanks during safing operation. Fill and drain loop can be used for backup; however, tank collapse would occur on re-entry	IIB
	Check Valve		
	1. Leakage	No effect - system shutoff valve in series prevents leakage flow between propellant tanks.	III
	2. Fails Closed	Loss of main tank pressurization purge capability for safing.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPELLANT TANK SAFING & VENTING

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Tank Purge & Blowdown for Safing (Continued)	APS Tank Vent/Fill Valve (LH_2 & LOX Tanks)	(Reference APS FMEA for operational failure modes)	
	1. Fails Closed	Unable to drain APS propellant tank. Inability to drain and purge propellant tank.	IIB
	Tug/Orbiter Vent Disconnect (GOX & GH_2)		
	1. Fails to Engage Properly/ Leakage	Propellant vapors would enter the cargo bay which can result in a hazardous condition to the orbiter crew.	IIB
	Tug/Orbiter Fill & Drain Disconnect (LH_2 & LOX)		
	1. Fails to Engage Properly/ Leakage	Propellant vapors would enter the cargo bay which can result in a hazardous condition to the orbiter crew.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPELLANT TANK SAFING & VENTING

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
He Pressure Source Tug/Shuttle Orbiter Interstage Adaptor (Propellant Tank & Compartment Purge)	He Receivers and Manifold		
	1. Rupture Potential	Unable to purge and pressurize propellant tanks for re-entry. Propellant tank collapse would occur during re-entry. Receiver rupture may cause damage to vehicle or orbiter.	IA
	Isolation Control Valve		
	1. Fails Closed	Unable to purge and pressurize propellant tanks and vehicle compartment for re-entry. Structural collapse would occur on re-entry.	IIB
	High Pressure Relief Valve(He)		
	1. Leakage	Loss of pressurization/purge capability. Unable to purge and pressurize propellant tanks for re-entry. Structural collapse would occur on re-entry.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPELLANT TANK SAFING & VENTING

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
He Pressure Source, Tug/Shuttle Orbiter Interstage Adaptor (Continued)	Mechanical Helium Disconnects		
	1. Failure to fully engage & seal (leakage)	Loss of pressurization/purge capability. Unable to purge and pressurize propellant tanks for re-entry. Structural collapse would occur on re-entry.	IIB
	Electrical Disconnect		
	1. Fails to Mate	Loss of pressurization/purge capability. Unable to purge and pressurize propellant tanks for re-entry. Structural collapse would occur on re-entry.	IIB
Insulation Purge System Pressure Control. Tug/Shuttle Orbiter Interstage Adaptor	Solenoid Shutoff Valve		
	1. Fails Closed	Loss of insulation purge system. Possible structural damage to vehicle during re-entry.	IIB
	Low Pressure Regulator		
	1. Fails to Regulate	Loss of insulation purge system. Possible structural damage to vehicle during re-entry.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPELLANT TANK SAFING & VENTING

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Insulation Purge System Pressure Control. Tug/Shuttle Orbiter Interstage Adaptor (Continued)	Low Pressure Relief Valve		
	1. Fails Open	Loss of He supply could be controlled by closing system solenoid shut-off valve. Loss of insulation purge system. Possible structural damage to vehicle during re-entry.	IIB
	2. Fails Closed	Inability to control system pressure subsequent to system regulator failure would result in structural damage to vehicle during re-entry (Second Order Failure).	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: PROPELLANT FEED SYSTEM (LOX & LH_2)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Fails to Supply Propellant (LOX or LH_2) to Main Engine.	Prevalve (LOX or LH_2). Fails to Open	Mission loss. Main engine cannot be started. APS is capable of sustaining a holding mode for 30 minutes for tug payload rescue. Thus assumes payload release capability if necessary for retrieval.	IIA
Fails to shut off propellant (LOX or LH_2) at Main Engine Shutdown	Prevalve (LOX or LH_2)Fails to Close	Main engine will be shut down by its main propellant valves. Prevalve failure mode will result in excess propellant boiloff when not isolated from engine by closed prevalves. Possible mission loss.	IIB
Ineffective Propellant (LOX or LH_2) Temperature Control	LH_2 Cooling Coil Valves Valves fail to Open Pressure Regulator Fails Closed	No LH_2 flow for system line and component cooling will result in excess propellant flow through main engine for pre-start chilldown. Also degraded performance of APS. Propellant loss may result in loss of mission capability, depending on time of failure.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: FUEL CELL THERMAL CONTROL SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Thermal Control	Freon - 21 Pump/Accumulator		
	1. Fails to provide adequate flow.	Thermal system degradation. Possible loss of tug mission objectives.	IIB
	Radiators (4 Req'd)		
	1. Structural Failure (Leakage of freon)	Loss of thermal coolant loop - fuel cell damage. Loss of tug mission objectives	IIB
Supplemental Cooling (H_2O)	H_2O Storage Tank		
	1. Internal Tank Leakage (Diaphragm)	Inability to pressurize H_2O tank will cause the loss of supplemental cooling of the fuel cell which is required during high electrical load requirements. Damage to fuel cell could result with possible loss of tug mission objectives.	IIB
	Helium Pressurization Control for H_2O Storage Tank.		
	1. Loss of Pressurization	Inability to pressurize H_2O tank will cause the loss of supplemental cooling of the fuel cell which is required during high electrical load requirements. Damage to	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: FUEL CELL THERMAL CONTROL SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Supplemental Cooling (H_2O) (Continued)	1. Loss of Pressurization (Continued)	fuel cell could result with possible loss of tug mission objectives.	

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: MAIN PROPULSION SYSTEM (MPS)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Fails to Provide Thrust	Main Engine Fails to Start	Loss of mission capability. NOTE: Detailed engine FMCA to be provided by engine supplier, which would determine engine component criticality and failure modes.	IIA
	Main Engine Explosion	Loss of Tug and Payload. Any engine component which will cause inadequate chill down (low mode) or other anomalies resulting in a hazardous condition during engine start or engine operation would be hazardous to the tug and payload.	IB
		Engine explosion due to inadequate purging and safing at the end of the mission, and after re-entry into orbiter bay, would result in a hazard to crew.	IA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: THRUST VECTOR CONTROL (TVC)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Thrust Vector Control	Actuator(2 Req'd) No Output, (Actuator Seizes) Motor/Pump. Low Output Inverter. No Output Servo Valve. (2 Req'd.) No Output Major Hyd Oil Leakage. (Any Component or Line)	Possible loss of mission. Engine would require shut down. Effect dependent on time of failure, degree of control lost, and detection. Possible control by APS as a backup system. Loss of hydraulic power will result in lock-up of TVC in the last commanded position.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: THRUST VECTOR CONTROL (TVC)

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Thrust Vector Control	Servo Valve (2 Required) Erroneous Output Resulting in Actuator Failing Hard Over or Excessive Response Thermal Control Failure Resulting in Sluggish or Erratic TVC.	Possible loss of mission. Engine may require shutdown, depending on time of failure, degree of failed condition and detection.	IIA

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: INSULATION PURGE SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Purge of LOX and/or LH_2 Tank Insulation	Selector Valve Fails to Operate. (Diverts Purge Gas to LOX and/or LH_2 Tank Insulation.	Possible launch delay. Purge of insulation only required during ground operations and prelaunch. No purge required after propellant loading.	III
	Solenoid Control Valve to LOX Tank Insulation Manifold Fails to Operate	Possible launch delay. Purge of insulation only required during ground operations and prelaunch. No purge required after propellant loading.	III
	Solenoid Control Valve to LH_2 Tank Insulation Manifold Fails to Operate	Possible launch delay. Purge of insulation only required during ground operations and prelaunch. No purge required after propellant loading.	III
Pressure Control of Tug Interior (External to Propellant Tanks)	Pressure Switch Fails to Actuate Upon Increasing Pressure Vent Valve Fails to Operate	No mission effect. Pressure switch is redundant. Vent valves are also redundant.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: INSULATION PURGE SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Pressure Control of Tug Interior (External to Propellant Tanks) (Continued)	Vent Valve Fails in the Open Position	Tug Interior Compartment Purge could not be maintained.	IIB

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: ORBITER/TUG DOCKING SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Orbiter/Tug Separation	Electric Actuator (actuates four latches) Fails to unlatch one or more latches. (4 places)	Mission loss. Tug fails to separate from orbiter. Tug must be returned to orbiter cargo bay, defueled and safed.	IIA
	Electric Actuated Latch (8 Required) One or more fails to unlatch	Mission loss. Tug fails to separate from orbiter. Tug must be returned to orbiter bay, defueled and safed.	IIA
Orbiter/Tug Docking for Re-entry	Probe/Drogue fails to engage and lock. (3 units)	Probe and drogue must engage and position tug for final latching operation. (8 latches). Two of the three probe/drogue operating successfully would still assure proper positioning and final alignment.	III
	Actuator/latch Fails to latch.	Possible tug structural damage if two or more adjacent latches are not engaged. Condition only affects latches subjected to tensional loads. Minimum of four latches equally spaced are capable of distributing the re-entry loads.	III
	(8 latches)	NOTE: See text for alternate methods that will be evaluated for improved reliability.	

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: TUG/PAYLOAD DOCKING SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Tug/Payload Separation	Latch Mechanism fails to unlatch. One (1) or more of 24 latches	Loss of mission. Tug payload cannot be released and placed in orbit. Remaining latches would have to be relatched and payload returned to orbiter.	IIA
Tug/Payload Docking (Payload Retrieval)	Probe and Drogue fails to engage and lock	Probe and drogue must engage and position payload for final latching operation. Successful operation of two (2) of the three (3) probe/drogue units would assure proper positioning and alignment	III
	Latching Mechanism fails to latch (24 provided)	Failure to latch two (2) or more latches would possibly damage tug and/or payload structure due to load concentration on remaining (engaged) latches. Condition only affects latches subjected to tensional loads. NOTE: See text for alternate methods that will be evaluated for improved reliability.	III

TABLE B-1. FAILURE MODE EFFECT ANALYSIS

SYSTEM: PROPELLANT MANAGEMENT SYSTEM

LOSS OF SYSTEM FUNCTION	COMPONENT-FAILURE MODE	MISSION EFFECT - REMARKS	CRIT. CAT.
Propellant Management (Propellant Mixture Ratio Control) During Main Engine Operation	Vehicle Main Computer		
	1. Fails to Provide Proper Engine Mixture Ratio Control Stimuli	Propellant consumption by the engine will not be made along the most efficient tank depletion schedule resulting in inefficient engine specific impulse and premature propellant depletion (LOX or LH_2)	IIA
	Discrete Level Point Sensor (LOX & LH_2)		
	1. Failure of Point Sensor to Provide Propellant Level Signal	Degraded system operation - each propellant tank will contain nine point sensors for liquid level measurement.	III
Propellant Loading	Capacitance Probe(LOX & LH_2)		
	1. Failure of Capacitance Probe to Provide Propellant Level Signal	Launch Delay - Each propellant tank contains a capacitance probe located in the upper portion of the tank	III